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THESIS

DEVELOPING EVALUATION MEASURES FOR THE SECOND STAGE NEXT GENERATION ENGINE ON EVOLVED EXPENDABLE LAUNCH VEHICLES

by

Jason A. Panczenko

March 2012

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The United States has been the leading nation in space technology, as space is a vital asset in military dominance. But to sustain its position in the area of space lift, the current U.S. second stage liquid propulsion engine, the RL10 (developed in 1958) needs to be replaced. This replacement requires systems engineering methods and new technological advances to adhere to mission requirements and constraints of current platforms. This thesis provides a history of the Evolved Expendable Launch Vehicle (EELV), U.S. liquid propulsion, and the RL10 LH2/LOX engine to analyze tradeoffs between major requirements in new upper stage development and to provide a recommendation of evaluation measures. The results are a proactive case presenting the benefits of a new upper stage engine on EELV, a tradeoff comparison between rocket propulsion engine cycles, a waterfall model for engine qualification and testing of liquid propulsion rocket engines, and testing recommendations for NGE qualification. Additionally, the thesis recommends specific impulse, thrust, and thrust-to-weight values that should be used as a design baseline for the next generation upper stage engine on EELV. These recommendations should be of value to engineers or program managers who are or will be responsible for acquiring replacement propulsion systems.

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ABSTRACT

The United States has been the leading nation in space technology, as space is a vital asset in military dominance. But to sustain its position in the area of space lift, the current U.S. second stage liquid propulsion engine, the RL10 (developed in 1958) needs to be replaced. This replacement requires systems engineering methods and new technological advances to adhere to mission requirements and constraints of current platforms. This thesis provides a history of the Evolved Expendable Launch Vehicle (EELV), U.S. liquid propulsion, and the RL10 LH2/LOX engine to analyze tradeoffs between major requirements in new upper stage development and to provide a recommendation of evaluation measures. The results are a proactive case presenting the benefits of a new upper stage engine on EELV, a tradeoff comparison between rocket propulsion engine cycles, a waterfall model for engine qualification and testing of liquid propulsion rocket engines, and testing recommendations for NGE qualification. Additionally, the thesis recommends specific impulse, thrust, and thrust-to-weight values that should be used as a design baseline for the next generation upper stage engine on EELV. These recommendations should be of value to engineers or program managers who are or will be responsible for acquiring replacement propulsion systems.

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LIST OF ACRONYMS AND ABBREVIATIONS

CDR	Critical Design Review
DCSS	Delta Cryogenic Second Stage
DoD	Department of Defense
EE	English Engineering
EELV	Evolved Expendable Launch Vehicle(s)
FRSC	Fuel Rich Staged Combustion
GAO	Government Accountability Office
ICBM	Intercontinental Ballistic Missile
ISP	Specific Impulse
JANNAF	Joint Army Navy NASA Air Force
LPRE	Liquid Propulsion Rocket Engine
LRSD	Launch and Range Systems Directorate
NASA	National Aeronautics and Space Administration
NGE	Next Generation Engine
NLF	National Launch Forecast
ORSC	Oxidizer Rich Staged Combustion
OSL	Office of Space Launch
PDR	Preliminary Design Review
PLF	Payload Fairing
RFI	Request for Information
SE	Systems Engineering
SI	International System
SpaceX	Space Exploration Technologies

SRR	Systems Requirements Review
TC	Thrust Chamber
ULA	United Launch Alliance
USAF	United States Air Force

EXECUTIVE SUMMARY

In the past 50 years, more than 40% of all historical launch vehicle failures were caused by propulsion subsystem malfunctions. A launch failure incurs a significant financial loss on an order of hundreds of millions to billions of dollars. These costs include but are not limited to launch hardware loss, payload loss, payload opportunity cost, and recovery operations. Reliability and sustainability of U.S. space lift will remain in question until the replacement of the current U.S. second stage liquid propulsion engine, the RL10, which was developed in 1958. Replacement of this engine must take full advantage of current systems engineering methods and new technological advances while adhering to requirements of the U.S. space lift mission and constraints of the current launch vehicle platforms. The primary boost systems for National Security Space in the United States are the Delta IV and Atlas V launch systems. Both of these systems, also collectively referred to as Evolved Expendable Launch Vehicles, use different variants of the RL10 engine for second stage propulsion. The RL10 engine has experienced issues with parts obsolescence, escalating cost due to low production rates, and outdated manufacturing methods. The Next Generation Engine will use current technology and manufacturing methods and have improved performance to enable Evolved Expendable Launch Vehicles sustainment to 2030, while at the same time maintaining the U.S. liquid propulsion industrial base and making the engine available for the next generation launch systems for Department of Defense and the National Aeronautics and Space Administration. This thesis addresses the tradeoffs between major evaluation measures for development of the Next Generation Engine on Evolved Expendable Launch Vehicles, recommends values for performance evaluation measures for the second stage Next Generation Engine, describes the benefits of a new second stage engine on Evolved Expendable Launch Vehicles, recommends an engine cycle for use on the next U.S. built upper stage, and addresses the testing methods for engine qualification.

In comparison, each engine cycle has potential benefits. The staged-combustion and the expander cycles achieve a higher specific impulse, thus are more efficient than, the gas generator cycle. However, the staged-combustion and expander cycles are also more complex and costly to develop. Up to this point the expander cycle has been the cycle of choice for upper stages. Calculations in this thesis have assumed an expander cycle. Currently due to a historical perspective and to stay in line with current Evolved Expendable Vehicles the author has recommended an expander cycle. However, the author suggests future research be done in order to consider the potential performance benefits of a staged-combustion cycle.

A systems engineering waterfall method is recommended by the author for use of both the acquisition process and testing of the Next Generation Engine. The first step in the systems engineering process is to identify the stakeholders. The primary stakeholders of the Next Generation Engine have been identified as the Launch and Range Systems Directorate (LRSD), NASA, the Office of Space Launch (OSL) and all of LRSD's primary customers. LRSD's primary customers include Defense Weather Systems Directorate, Space-Based Infrared Systems Directorate, Global Positioning Systems Directorate, MILSATCOM Systems Directorate, and the U.S. Navy. It is a recommendation of the author that Air Force Evolved Expendable Launch Vehicle systems engineers explore other possible primary stakeholders throughout the early acquisition process.

Evaluation measures development using analytical relationships among historic engine data and regression methods found an optimal specific impulse of 463 seconds (sec) and thrust of 32992 pound force (lbf) for Evolved Expendable Launch Vehicles. The author recommends performance evaluation measures of 463sec or higher specific impulse and 29000–33000lbf thrust for the NGE, while maintaining an engine thrust-to-weight ratio of 35 or higher.

Test evaluation measures for engine development vary drastically with each engine program. The author recommends the use of and has highlighted testing evaluation measures presented in Reference 18**,** Test and Evaluation Guideline for Liquid Rocket Engines, by the Joint Army Navy NASA Air Force Assembled Team. The author also recommends the continuation of the test-like-you-fly method.

The test-like-you-fly methodology tests flight hardware in environments and operating conditions experienced during first flight conditions. When testing cannot be performed at these conditions ample margin is added.

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I. INTRODUCTION

In the past 50 years, more than 40% of all historical launch vehicle failures were caused by propulsion subsystem malfunctions. A launch failure incurs a significant financial loss on an order of hundreds of millions to billions of dollars. These costs include but are not limited to launch hardware loss, payload loss, payload opportunity cost, and recovery operations. Experts advise that the key to mitigating launch failure is to improve the reliability of launch vehicle propulsion subsystems. The primary boost systems for National Security Space in the United States are the Delta IV and Atlas V launch systems. Both of these systems, also collectively referred to as Evolved Expendable Launch Vehicles (EELV), use different variants of the RL10 engine for 2nd stage propulsion. The RL10 was originally designed in 1958. The RL10 engine has experienced issues with parts obsolescence, escalating cost due to low production rates, and outdated manufacturing methods. The Next Generation Engine (NGE) will use current technology and manufacturing methods and have improved performance to enable EELV sustainment to 2030, while at the same time maintaining the U.S. liquid propulsion industrial base and making the engine available for the next generation launch systems for Department of Defense (DoD) and the National Aeronautics and Space Administration (NASA). This thesis will help to define the evaluation measures desired for the NGE. The thesis will also make recommendations on engine cycles and qualification evaluation measures for new second stage propulsion.

A. SCOPE

This thesis will focus on defining evaluation measures for the NGE using a trade study approach and regression analyses based on vehicle performance simulation results for the Atlas V and Delta IV EELV launch vehicles. The National Launch Forecast (NLF) will be used to define the future mission requirements for EELV. Representative vehicle performance simulations will be setup using the RL10 engine performance with a generalized mission trajectory as a baseline. Values for NGE thrust, specific impulse, mixture ratio, and weight will be defined based on conceptual designs and engine power cycle capabilities. The vehicle performance simulations will be run with the range of NGE parameters using a Design of Experiments approach. Further trade studies will be performed and will define a recommendation for propulsion cycle usage. The research conducted for this thesis indicates that the Air Force's interests will be met if the application of SE process models, such as the waterfall method, were used for testing processes and qualification.

B. PURPOSE

The purpose of this thesis is to develop and perform a tradeoff analysis that will identify and develop performance evaluation measures needed for the next generation second stage engine on Evolved Expendable Launch Vehicles. This analysis work will be used by the NGE systems engineer as a tool in the future acquisition of the next upper stage engine on EELV.

C. RESEARCH

This research will utilize systems engineering processes and tools in order to develop evaluation measures for new liquid propulsion design. Research topics include liquid propulsion system performance characteristics, engine cycles, and development criteria. Primary methods of research include tradeoff studies among engine cycles and evaluation measures accomplished by evaluating trajectory and mission simulation results. Research questions that will be addressed include but are not limited to:

1. What are the performance tradeoffs between thrust and specific impulse? What connections among these performances factors and others?
2. What are the important evaluation measures for a baseline second stage EELV mission?
3. What are the benefits a new 2nd stage can bring to EELV?
4. Which engine cycle may be the most beneficial for use on the NGE?
5. What are the test methods for engine qualification?

D. BENEFITS OF STUDY

This thesis will benefit the liquid propulsion community, the EELV program, and launch programs in general. It will also explore key performance parameters for upper stage engine design as they relate to EELV. Trade study and sensitivity study techniques will be developed and applied to formulate a practical recommendation. The thesis will also provide a basis for the development and procurement of the NGE. The NGE will replace the EELV upper stage propulsion and is a candidate engine for the NASA Space Launch System upper stage.

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II. EVOLVED EXPENDABLE LAUNCH VEHICLE HISTORY

United States' liquid rocket propulsion technology has evolved over the past five decades to meet the changing needs of the commercial community, agree with DoD diminishing budgets, and ensure national access to space (Coleman, 2000). The importance of the United States having assured access to space is best laid out by the National Presidential Directive Number 40 which states:

> Access to space through U.S. space transportation capabilities is essential to: (1) place critical United States Government assets and capabilities into space; (2) augment space-based capabilities in a timely manner in the event of increased operational needs or minimize disruptions due to on-orbit satellite failures, launch failures, or deliberate actions against U.S. space assets; and (3) support government and commercial human space flight. The United States, therefore, must maintain robust, responsive, and resilient U.S. space transportation capabilities to assure access to space ("Nspd-40:u.s space transportation, 2004).

The United States' current solution to assure space access for operational space assets is to maintain two families of launch vehicles under the Evolved Expendable Launch Vehicle (EELV) program.

The EELV program was initiated in 1995 as the Air Force's premium space lift modernization program. The purpose of this program was to reduce the cost of operational space launch by 25–50% and to improve reliability over the heritage launch systems (Atlas II, Delta II, and Titan IV). Procurement of EELV boosters for military space launch was to evolve into a "commercial like" nature (Buzzatto, 2003). The EELV program eventually produced two families of launch vehicles as the solution to U.S. space lift needs. These two families are the Delta IV launch system, developed by McDonnell Douglas (now The Boeing Company), and the Atlas V launch system, developed by Lockheed Martin. See Figure 1 for a depiction of the EELV launchers.

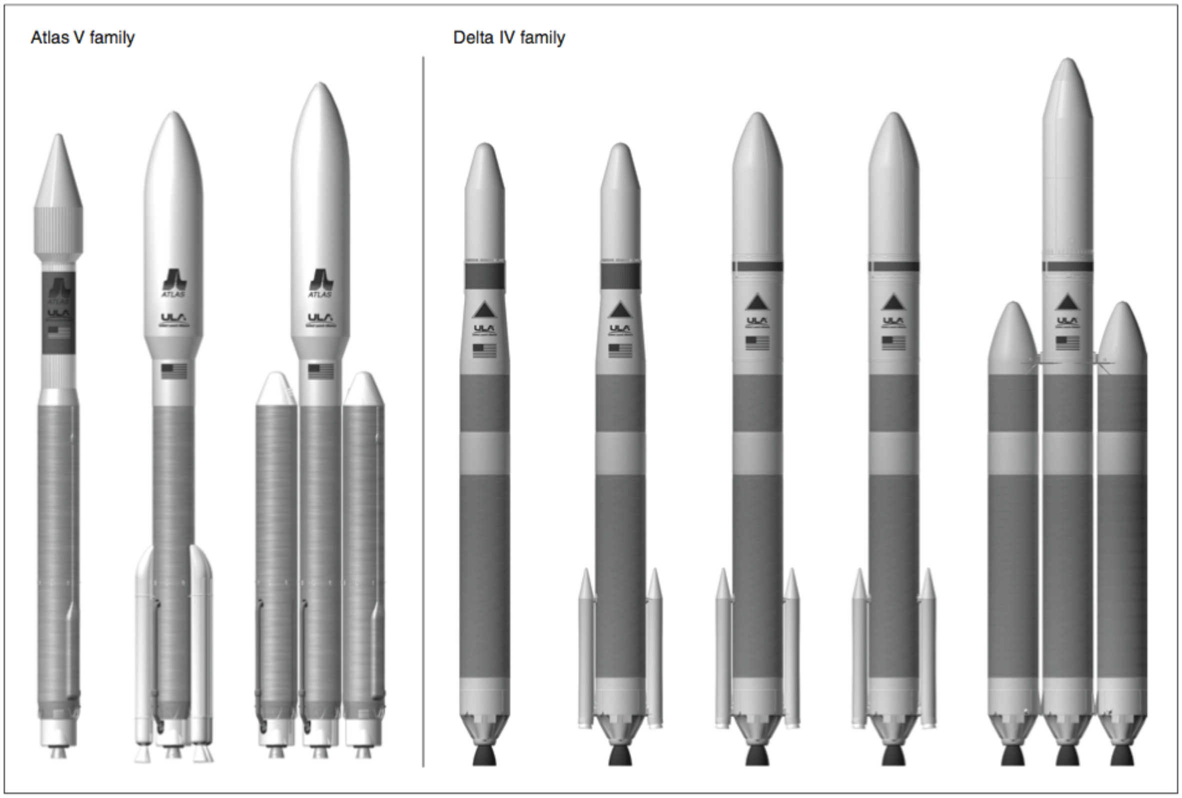

Figure 1. The Atlas V and Delta IV Space Launch Boosters Make Up EELV (From: GAO, 2008)

The United States Air Force (USAF) oversaw the development of the two new launch systems in just five years. (Buzzatto, 2003) Boeing's Delta IV and Lockheed Martin's Atlas V programs eventually merged to form the United Launch Alliance (ULA).

A. CURRENT BOOSTER SYSTEMS

The Atlas V and Delta IV systems evolved through different paths. Major differences include launch processing, launch pad operational concept, and some major components.

1. Atlas V

The Atlas V space launch system has a lineage which began in 1954 under the ICBM program (Minami, 1991).

The first Intercontinental Ballistic Missile (ICBM) was the SM-65 Atlas. Although the Atlas started as ICBM technology, emergence as a space launch vehicle (SLV) followed soon thereafter (Minami, 1991). Figure 2 illustrates the heritage of the Atlas system.

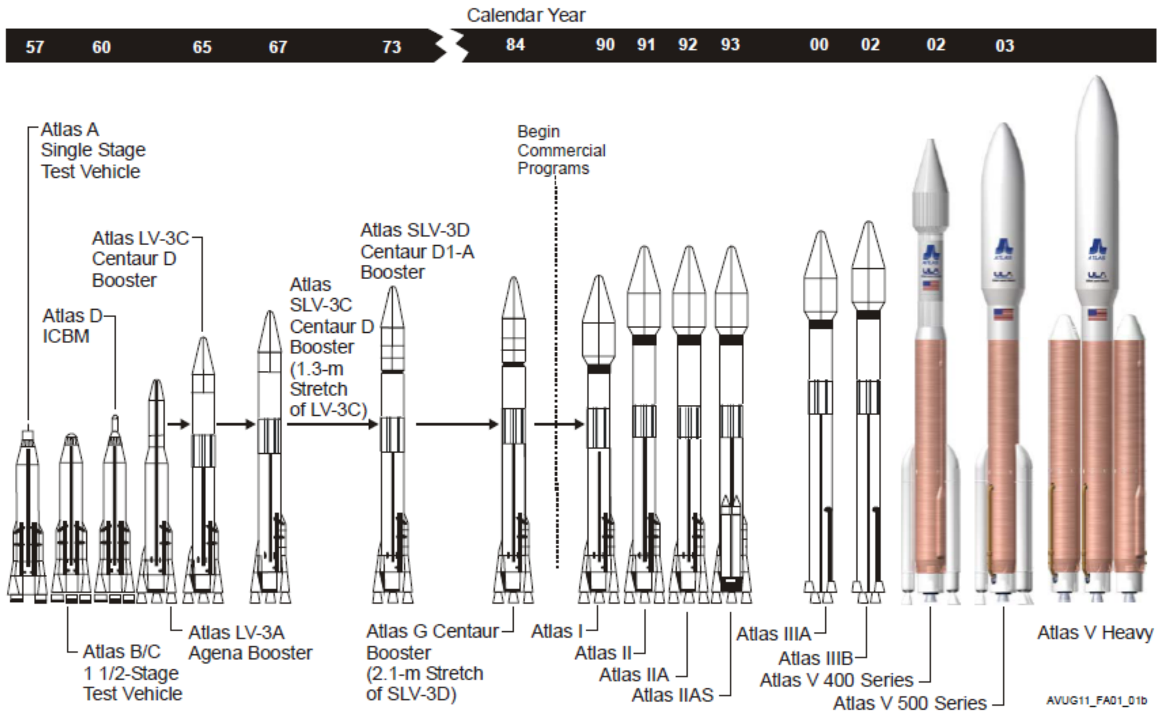

Figure 2. Heritage of the Atlas ICBM/SLV (From: Atlas V Users' Guide)

a. Atlas V Configuration

Main features, as seen below in Figure 3, include the Common Core Booster™ powered by a RD AMROSS RD-180 engine, Aerojet strap-on solid boosters (up to five), Centaur upper stage powered by a single or dual Pratt and Whitney Rocketdyne RL10A-4-2 engine(s), and an option of a 4.2 or 5.4 meter payload fairing (PLF) ("Atlas V launch," 2010). A three-digit (XYZ) naming convention is used for the Atlas V configuration identification. The first digit represents the payload fairing size (either 4 or 5), the second digit represents the number of solid rocket boosters used (1 through 5), and the third digit represents the number of engines used on the Centaur (1 or 2)

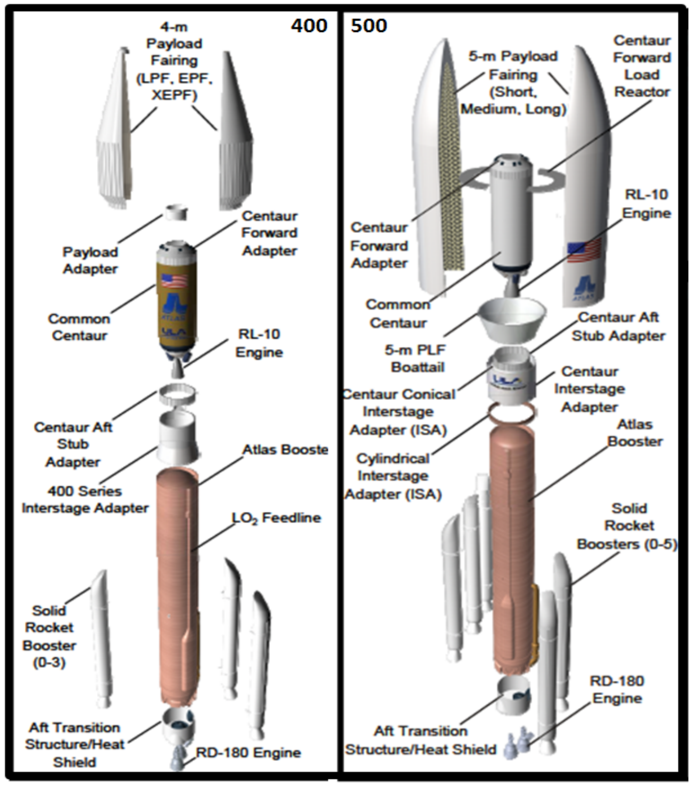

Figure 3. Atlas V 400 & 500 Series Configurations (From: Atlas V Users' Guide)

b. Atlas V Performance

The performance of the Atlas V is depicted in Table 1. Due to missions already being on contract with the Atlas V rocket, it is vital that the Next Generation Engine not degrade the current performance. This will become more apart during the analysis portion of this thesis.

PERFORMANCE

	401	431	551	HLV
GTO	4,750 kg (10,470 lb)	7,700 kg (16,970 lb)	8,900 kg (19,260 lb)	13,000 kg (28,660 lb)
LEO	9,370 kg (20,650 lb)	15,130 kg (33,650 lb)	18,510 kg (40,800 lb)	29,400 kg (64,820 lb)

Table 1. Weight to Orbit Performance of the Atlas V (From Atlas V Users' Guide)

2. Delta IV

The Delta SLV is a direct descendant of the Thor missile. Originating as a launch vehicle in the 1950s, the Delta program was initiated by NASA. Figure 4 shows the lineage of the Delta space launch system ("Delta payload planners," 2007).

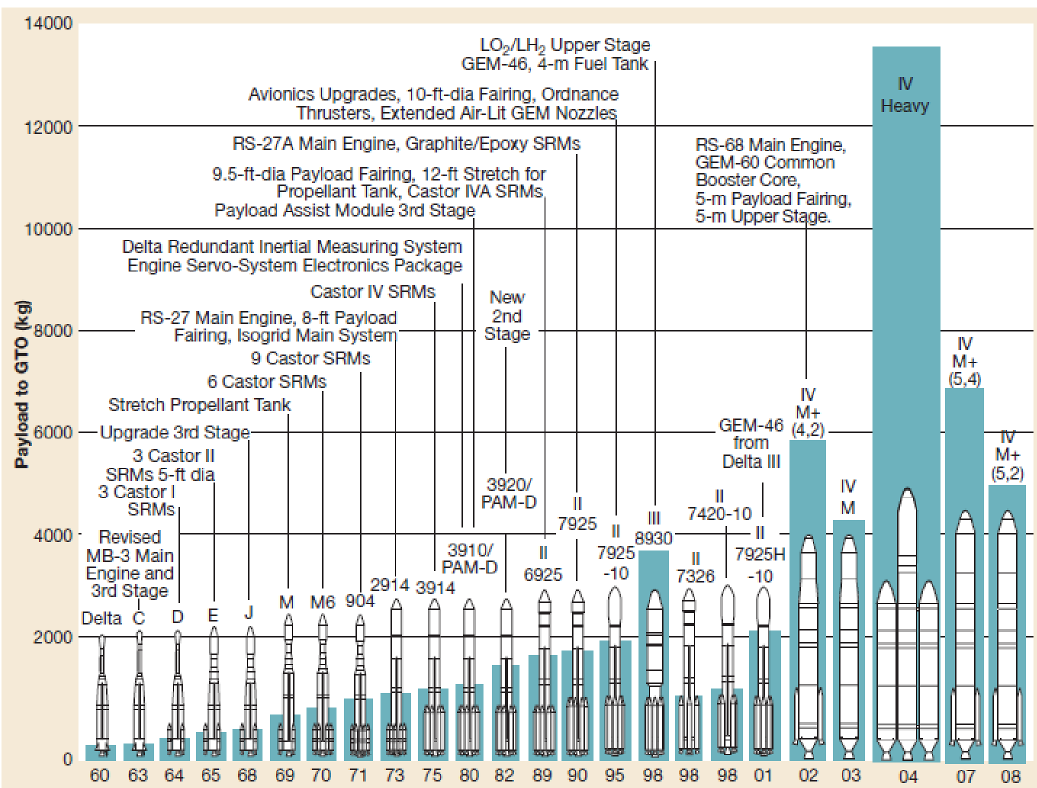

Figure 4. Heritage of the Delta SLV (From: Delta Payload Planners Guide)

a. Delta IV Configuration

Main features, as seen below in Figure 5, include the Common Booster Core powered by a Pratt and Whitney Rocketdyne RS-68 engine, a Delta Cryogenic Second Stage (DCSS), and off pad horizontal vehicle integration. There are three variants of Delta IV M+ configuration. The Delta IV M+(4,2) uses two strap-on solid rocket motors (SRMs) to augment the first-stage CBC and a 4-m diameter DCSS and PLF. The Delta IV M+(5,2) and Delta IV M+(5,4) have two and four SRMs, respectively and 5-m-diameter DCSS and PLF. The Heavy Lift Vehicle (HLV) variant has two strap-on CBC cores with a 5-m DCSS and PLF ("Delta payload planners," 2007) .

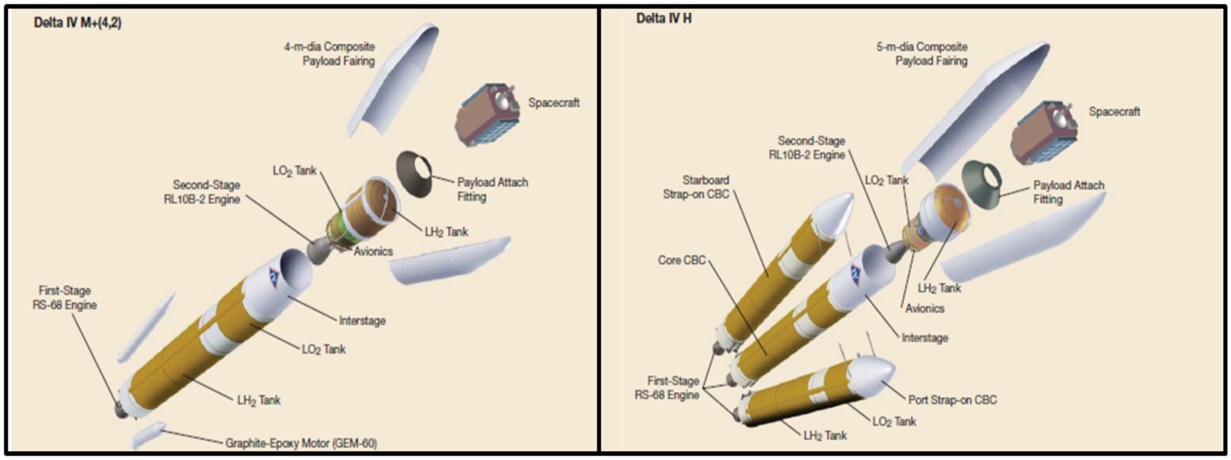

Figure 5. Delta IV M+(4,2) and HLV Configuration Expanded View (From: Delta Payload Planners Guide)

b. Delta IV Performance

The performance of the Atlas V is depicted in Table 2, below. Due to missions already being on contract with the Delta IV rocket, it is necessary that the Next Generation Engine not degrade the current performance.

PERFORMANCE

	Medium	M+(4,2)	M+(5,4)	Heavy
GTO	4,300 kg (9,480 lb)	6,030 kg (13,290 lb)	7,020 kg (15,470 lb)	12,980 kg (28,620 lb)
LEO	9,150 kg (20,170 lb)	12,240 kg (26,980 lb)	13,360 kg (29,450 lb)	22,560 kg (49,740 lb)

Table 2. Weight to Orbit Performance of the Delta IV (From: Delta Payload Planners Guide)

B. RL10 BACKGROUND

The Atlas V and Delta IV rocket systems vary considerably in many areas. However, these two EELV boosters do have one significant similarity. The Centaur of the Atlas V and Delta IV Cryogenic Second Stage, both use variants of the Pratt and Whitney Rocketdyne RL10. See Figure 6 for an illustrated comparison.

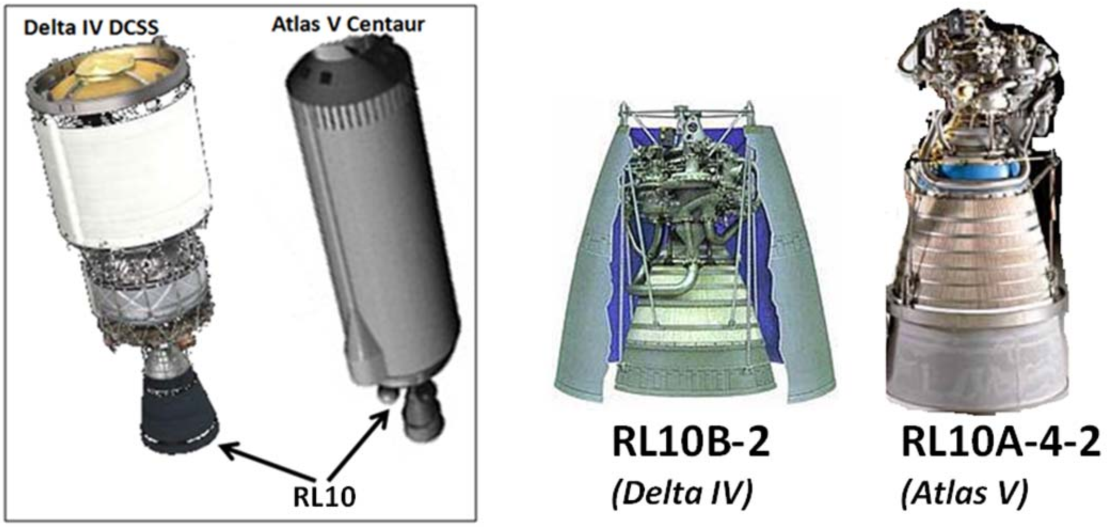

Figure 6. EELV upper stage comparison (left); illustration of the RL10 Variants (right) (From: Cooley, 2009)

The RL10 family of engines are multi-start, expander cycle engines. Some of engines of the RL10 family are capable of mixture ratio control to maximize vehicle propellant usage (Emdee, Fentress & Malinowski, 1997). The advantages to the expander cycle are addressed in Chapter III, Section b (Engine Cycles) of this thesis.

The RL10 was the first rocket engine to use liquid hydrogen and liquid oxygen (LOX) propellants (Wiswell & Huggins, 1990). No other U.S. in-space propulsion system has flown using this propellant combination (Sackheim, 2006). This is rather impressive considering the RL10 had its inception in 1958.

The Delta IV RL10B-2 includes the world's largest carbon-carbon extendible nozzle. The dimensions of the nozzle enable a high expansion ratio and high specific impulse of 465.5 seconds in a vacuum (Cooley, 2009). See Table 3 for characteristics of the RL10A-4-2 and the RL10B-2. The RL10B-2 runs a constant mixture ratio (*MR*) of 5.88. Considering the RL10B-2 *MR* is fixed, fuel levels before liftoff are adjusted accordingly for each mission. The advertised vacuum specific impulse of the RL10 is different than the EELV flight correlated specific impulse. This thesis uses the flight correlated specific impulse for any and all calculations made or analysis accomplished.

Characteristics	
	RL10A-4-1/RL10A-4-2
Thrust:	22,300 lb
Weight:	370 lb
Fuel/oxidizer:	Liquid hydrogen/Liquid oxygen
Mixture ratio:	5.5:1
Specific impulse:	451.0 sec

RL10B-2 Characteristics	
Thrust:	24,750 lb
Weight:	664 lb
Fuel:	Liquid hydrogen
Oxidizer:	Liquid oxygen
Mixture Ratio:	5.88:1
Specific Impulse:	465.5 sec
Length (stowed): (deployed):	86.5" 163.5"
Diameter (nozzle extension):	84.5"

Table 3. RL10B-2 & RL10A-4-2 Characteristics Comparison (After: Cooley, 2009)

The Atlas V RL10A-4-2 does not include the extendable nozzle the RL10B-2 has. This makes the RL10A-4-2 engine 294lbs lighter than the RL10B-2. However, without the extendable nozzle, the expansion ratio of the RL10A-4-2 is much lower than the RL10B-2. With a lower expansion ratio, the RL10A-4-2 has a lower specific impulse than the RL10B-2. The RL10A-4-2, unlike the RL10B-2, has the ability to vary its mixture ratio.

C. REASONS FOR A NEW UPPER STAGE ENGINE

Although the RL10 has been a strong performer for over 50 years, the current operating conditions have eroded the design margin for some components. With improvements of a 100% increase in chamber pressure and 30% increase in turbine speed since engine inception, the RL10 performance margins have been reduced.

The commercial launch demand for EELV is low when compared to the original forecasts ("Uncertainties," 2008).

The cancellations of both the NASA Constellation and the Space Shuttle programs have decreased the demand for upper stage propulsion even further. Low commercial demand for EELV has led to low production rates and escalating cost growth (Chase, 1995).

The RL10 manufacturing methods are still that of over 50 years ago. Aged manufacturing methods are labor intensive and have low yields. Slowly, over time, parts and materials of the RL10 become increasingly closer to obsolesce. Many of the manufacturing techniques require the skills of master craftsmen. These skills are often difficult to find today.

While the rest of the world over the last 40 years has developed an estimated 40+ new rocket engines, U.S. new rocket engine development has been limited since 1988. Until the recent in-space flight of the Space X Merlin LOX/RP-1 liquid propulsion rocket engine (LPRE), the RL10 was the only U.S. in-space LPRE that used cryogenic propellants. The Merlin engine is immature and was not designed as an in-space specific LPRE (Sackheim, 2006). See Figure 7 for a graphical depiction of major rocket engine development in the U.S.

All of the considerations discussed provide rationale for a new upper stage engine for EELV to improve performance margins, support a lagging industrial base, update manufacturing processes to reduce cost, and to maintain U.S. LPRE expertise.

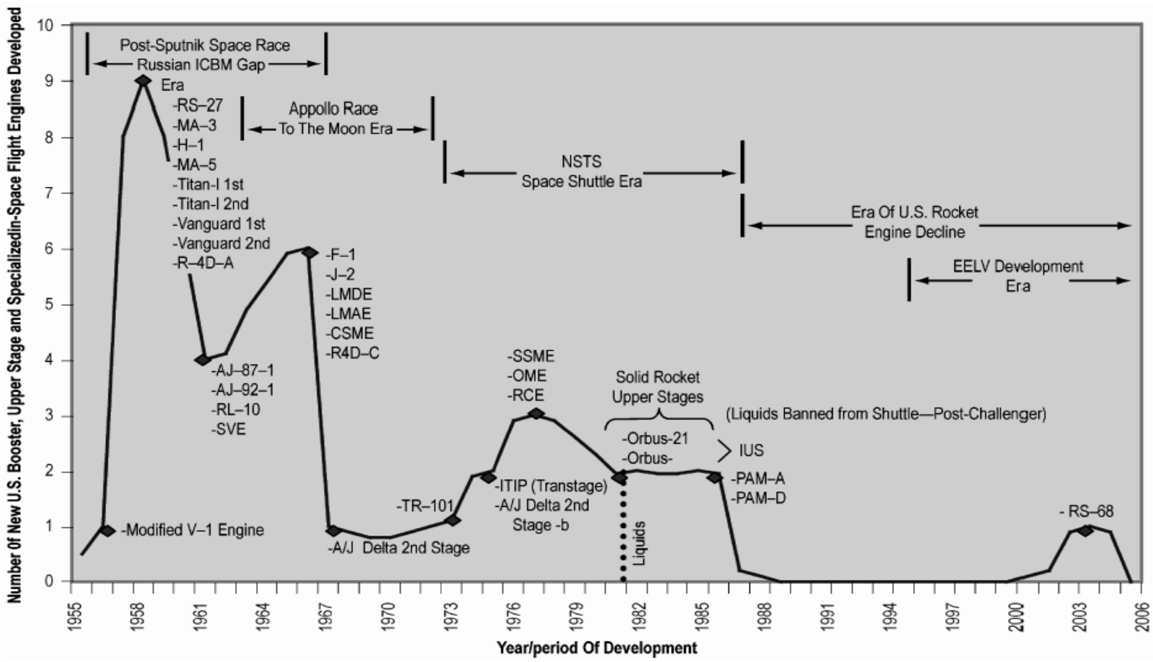

Figure 7. U.S. Rocket Developments From 1955–2005 (After: Sackheim, 2006)

*Note: Not shown here; the Merlin LPRE 1st flight was in 2006

D. FUTURE OF EELV

Current plans for an EELV follow-on are slated for 2030 or later. Technologies currently in development are unlikely to have a significant impact on either risk or cost if implemented on a potential EELV follow-on. With consideration to the immense cost of developing a follow-on to EELV, it is unlikely any new system/family is anywhere in our near future (Sackheim, 2006). Sustainment of EELV will be vital to having assured access to space.

E. CHAPTER SUMMARY

United States' liquid rocket propulsion technology today is based on commercial need, DoD budgets, and the DoD policy of assured national space access. The current solution to U.S. space lift need is the EELV systems, the Atlas V and Delta IV. Both EELV systems today use variants of the RL10 liquid propulsion engine. The RL10 engine has been in service for over 50 years. The age of the RL10 has created issues with old manufacturing methods and parts obsolescence. The current low production rates of the RL10 have an impact of escalating cost growth. Lastly, improvements to the RL10 during its life cycle have increased performance output while lowering reliability margins.

New liquid propellant rocket engine development in the U.S. has been stagnant when compared with the rest of the world. The U.S. will need to develop in-space LPRE technology in order to maintain an industrial base and remain a dominant space power.

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III. BACKGROUND AND FUNDAMETALS OF LIQUID PROPULSION

The technological level in the field of liquid propulsion is very mature. During the 1940s and 1950s the basic construct and major components were defined. LPREs in production today have achieved high reliability. Goals of LPREs today are high performance, steadfast reliability, high safety and low cost production (Sutton, 2003). Future LPREs will look at reusability to save on cost.

Although every LPRE has the same major structure, LPREs vary significantly by mission. All liquid propulsion engines have one or more thrust chambers, a feed system for providing a propellant to the thrust chamber, and a control system. Differences occur in LPREs between high or low thrust, cryogenic versus storable propellants, monopropellants vs. bipropellants, single use or reusable, and variable or constant thrust. Other differences among LPREs include the number of restarts and engine cycle (Sutton, 2003).

This thesis focuses on bipropellant liquid propulsion upper stage engines. These engines use a fuel and oxidizer in a liquid form. Upper stage LPREs will differ from many others in that they need to be designed to operate outside of our Earth’s atmosphere. Please see Figure 8 to identify the propulsion area of focus.

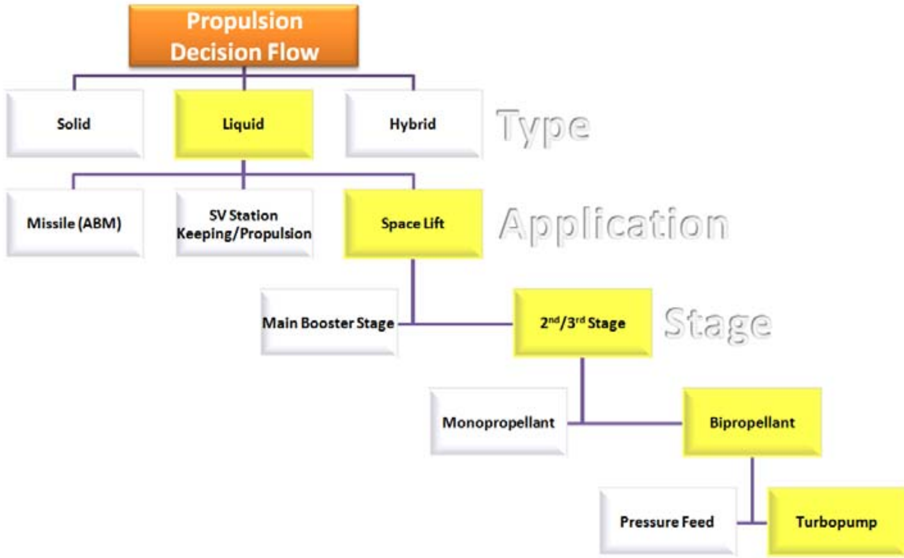

Figure 8. High Level Propulsion Application Decision Flow

A. HISTORY OF LIQUID PROPULSION IN THE UNITED STATES

The history of U.S. liquid propulsion starts with one American, Robert H. Goddard. Robert Goddard's legendary rocket experimentation provided many firsts for the world and earned him the distinction as "The Father of Modern Rocketry." Sutton (2003) provides an in depth description of innovations produced by the United States in the history of rocket development.

> There were a good number of inventions, innovations, or first implementations of technology that can be credited to organizations in the United States. This includes the first flight of an engine with liquid hydrogen as a fuel, the first expander engine cycle, the theory of bell-shaped nozzles, the first turbo pump, the first booster pump, and the first applications of gimbals in thrust chambers for vector control. Furthermore, there was the development of the tubular thrust chambers for regenerative cooling, the first use of ablatives on LPREs, the development of special materials for the hot-TC walls and for turbine buckets, flat plate machined injectors made of forgings, certain clever valve designs, certain injection patterns, Aerojet's platelet injectors, the electronic engine controls, pressurizing propellant tanks with gas generators, variable position pintle injectors, first flight of an extendible nozzle, first application of liquid side injection for TVC, and very fast small propellant valves mounted on an injector of a small thruster (Sutton, 2003).

A brief historical timeline of liquid propulsion in the United States can be seen in Figure 9.

History of Liquid Propulsion in the United States

Year	Event
1919	- Robert H. Goddard publishes his famous paper "A Method of Reaching Extreme Altitudes"
1921-1923	- Goddard develops first high-pressure gas feed system
1926	- Goddard flies the world's first rocket with a liquid propulsion rocket engine
1930	- Goddard is the first to use baffles to suppress sloshing
1935	- Guggenheim Aeronautical Laboratory of California Institute of Technology (GALCIT) started rocket research [GALCIT would eventually become NASA's Jet Propulsion Lab
1938	- Regenerative cooling first demonstrated In the US by James H. Wyld
1940	- Turbopump and Gas Generator flown in a sounding rocket by Goddard
1943	- Goddard tested the first US variable thrust LPRE
1943	- RMI 6000-C4 LPRE propelled the Bell X-1 to a record spped of Mach 1.06
1951	- Bomar Area Defense System (ramjet-powered superonic missile) developed by Aerojet
1953	- US Army Redstone MRBM (Rocketdyne) first flight; the Redstone SLV would eventually run suborbital missions for the Mercury program
1956	- Navaho cruise missile (Rocketdyne) first flight
1957	- Beginning of the Atlas ICMB/SLV family, first flight of the Atlas ICBM (Rocketdyne) (Atlas was the 1st USAF ICBM); the Atlas SLV was used for orbital missions of the Mercury program
1957	- Thor/Delta launch family begin with the first flight of the Thor IRMB (Thor was the 1st USAF IRBM)
1957	- Jupiter MRBM (Rocketdyne) first flight (later Juno SLV was derived from Jupiter)
1961	- Saturn I first flight (Rocketdyne H-1 LPRE); eventual follow on of the Saturn V (using Rocketdyne F-1 and J-2 LPREs) in 1967; Saturn V was used in the Apollo program
1963	- AIAA was formed
1963	- First flight of the RL-10 (Pratt and Whitney) flown, 1st pump fed LOX/LH2 LPRE // Accredited today with the highest specific impulse (467s)//1st LPRE to use the expander cycle
1963	- Titan II ICBM (Aerojet) became operational // later Titan III & IV follow ons would be uprated SLVs with larger engines for heavy space lift, Titan II SLVs carried astronaunts durin the Gemini program
1981	- Space Shuttle first flight, using the SSME (Rocketdyne) (SSME is the only US stage-combustion cycle engine)
1993	- Pratt and Whitney reach agreement with NPO Energomash to license/eventually produce Russian LPRE; includes RD-170, RD-180, & RD-120
2002	- First flight of the RS-68 (Rocketdyne) on the EELV Delta IV, the first LPRE to be fully designed by computers
2006	- Space-X Merlin LPRE first flight, 1st privately-developed LPRE to reach orbit

Figure 9. History of Liquid Propulsion in the United States (After: Sutton, 2003; Sackheim, 2006; and Spires, 1998)

B. ROCKET PERFORMANCE FUNDAMENTALS

The foundation of rocketry lies in accordance with Newton's Third Law of Motion and the fact that, unlike other propulsion systems, a rocket carries everything it needs to operate without external oxygen (Goddard, 1919). Underlying rocket principles are rooted in physics, fluid mechanics, thermodynamics, and chemistry.

1. Definitions and Equations

It is essential to outline basic rocketry equations and define common terms for a general understanding of analysis of Chapter IV. To develop understanding and knowledge of propulsion beyond this thesis scope see Reference 19.

a. *Specific Impulse*

The total impulse I_t is the thrust force F integrated over the duration of the rocket engine burn time:

$$I_t = \int_0^t F dt \tag{3-1}$$

Note, the total impulse I_t is proportional to the total energy released by all the propellant in a propulsion system (Sutton & Biblarz, 2010).

The specific impulse I_{sp} is an important measure of rocket engine performance. Specific impulse can be defined as the total impulse per unit of weight of propellant (Sutton & Biblarz, 2010). Essentially, the higher the specific impulse, the better the performance of the rocket engine. To get a time-averaged specific impulse with a total mass flow rate $\dot{m}$ and an acceleration of gravity g the following equation is used:

$$I_{sp} = \frac{\int_0^t F\, dt}{g \int \dot{m}\, dt} \tag{3-2}$$

The specific impulse is often seen in a simplified version. If we assume a constant mass flow rate, a constant thrust, and negligible start/stop transients then Equation 3-2 will reduce to the following:

$$I_{sp} = F / (\dot{m} g_0) = F / \dot{w}$$
$$I_t / (m_p g_0) = I_t / w \tag{3-3}$$

This equation uses w as a substitute for total propellant weight ($m_p g_0$) and $\dot{w}$ is in reference to the weight flow rate (Sutton & Biblarz, 2010).

The units for specific impulse in both the International System (SI) and the English Engineering (EE) are seconds. Be mindful that seconds in this particular case are not a measure of time but rather a measure of thrust per unit flow rate. Specific impulse, as stated above, leads to drastic performance improvements with even slight increases. To achieve a mere one second improvement in specific impulse is significant to the system.

b. Performance and Efficiency

In analysis of rocket vehicle flight performance both the mass ratio MR and the propellant mass fraction ζ are important values. The MR is heavily influenced by material choices (McGinnis & Einspruch, 2006). MR for a rocket system is defined by the initial mass m_0 and the final mass exclusive to the system m_f. The m_0 of the system can be further defined by

$$m_0 = m_f + m_p \tag{3-4}$$

where the m_p used above refers to the mass of the propellant of the system. Now given Equation 3-4 the MR relationship can be seen as

$$\mathsf{MR} = m_f / m_0 \tag{3-5}$$

The propellant mass fraction has been used as an indication of quality. It can be defined using the equation below (Sutton & Biblarz, 2010).

$$\zeta = m_p / m_o \tag{3-6}$$

Correlations between the above equations regarding flight performance as well as more information on flight performance relationships can be reviewed in Chapter 4 of Reference 19 (Sutton & Biblarz, 2010).

The characteristic velocity c^* relates the efficiency of combustion but is independent of nozzle characteristics. This value is often used in propulsion system performance comparisons (Sutton & Biblarz, 2010). The equation for characteristic velocity is

$$\text{—} \tag{3-7}$$

For Equation 3-7 the pressure inside the thrust chamber is referred to as p_c and the area at the nozzle throat is referred to as A_t. It can be shown that c* is inversely related to the molecular weight of the combustion product and hence a strong function of the propellant selected. A low molecular weight propellant like hydrogen will have a higher c* than kerosene.

For a given propellant, the oxidizer to fuel mixture ratio, *MR*, has a direct relationship with engine performance. The *MR* will impact the combustion product molecular weight and c* as noted above. In addition, while designing the plumbing and tanks of the propulsion system, the *MR* is a key parameter (Kim, Emdee, & Cohn, 2010). Considering the correlation between the plumbing/tanks of the propulsion system and *MR*, the NGE will be constrained in the *MR* range that is possible. The mixture ratio is defined as

$$MR = \frac{\dot{m}_{oxidizer}}{\dot{m}_{fuel}} \tag{3-8}$$

"The performance of gas expansion through a nozzle is quantified using a dimensionless parameter called the thrust coefficient C_f by comparing it to the thrust over the throat area," (Kim, Emdee, & Cohn, 2010). For engines operating in space, a large nozzle expansion ratio will result in a higher thrust and thus a larger C_f. C_f can be expressed as:

$$C_f = \frac{F}{A_t P_c} \tag{3-9}$$

c. Thrust

"Thrust is the primary reaction force experienced by a structure because of ejection of high-velocity exhaust gases," (Kim, Emdee, & Cohn, 2010). Using the law of conservation of momentum, the equation for thrust *F*

$$F = \dot{m}v_e + (p_e - p_a)A_e \tag{3-10}$$

can be seen as an addition of momentum thrust, $\dot{m}v_e$, and pressure thrust, $(p_e - p_a)A_e$. Respectively, p_e refers to the pressure at the nozzle exit, p_a refers to the atmospheric pressure, A_e refers to the area at the nozzle exit, and v_e refers to the exit velocity of the gas leaving the nozzle. The exit velocity is influenced by *MR* and is a direct function of the expansion ratio, A_e/A_t. Thus Equation 3-3 and 3-9 show that engine performance (thrust and specific impulse) increases with increased expansion ratio and will vary with *MR*.

"The thrust-to-weight ratio F/w_0 [often expressed as T/W] expresses the acceleration that the engine is capable of giving to its own loaded propulsion system mass," (Sutton & Biblarz, 2010). This value is often used in propulsion literature to compare the engine weight efficiency of different types of rocket engines. For upper stages, the engine is carried to orbit with the payload; therefore a higher engine T/W increases the payload capability of a rocket.

d. Relationships

For the purpose of fully understanding the analysis portion of this thesis, it is important to describe the interconnection between performance measurements.

By rearranging the thrust coefficient Equation 3-9, one can form a new equation for thrust.

$$F = C_f A_t P_c \tag{3-11}$$

Substituting this new thrust equation into the original equation for specific impulse, Equation 3-3, one can develop an equation for specific impulse that is a function of the thrust coefficient.

$$I_{sp} = \frac{C_f A_t P_c}{\dot{w}} \tag{3-12}$$

If equations 3-7, and 3-12 are combined, a new equation for specific impulse can be formed as Equation 3-13 below.

$$I_{sp} = \frac{C_f c^*}{g} \tag{3-13}$$

This equation shows that specific impulse is a function of the nozzle expansion ratio, embodied in C_f, and the selected propellant choice and *MR* (i.e., molecular weight) as embodied in c*. Since specific impulse is a measure of the efficiency of the engine, a more efficient engine will have a large expansion ratio and utilize low molecular weight combustion products like hydrogen.

Finally, it is important to understand the benefit of engine thrust during rocket ascent and during in-space operations. Using Newton's Second Law of Motion and accounting for the external forces on a rocket, an equation for the acceleration of a rocket body can be created:

$$a = \frac{F_t - F_d - F_g}{m}$$

$$given: \; F_g = mg \qquad a = \frac{F_t - F_d - mg}{m} \tag{3-14}$$

In Equation 3-14, F_t refers to thrust force, F_d refers to the drag force and F_g refers to the gravitational force. Equation 3-14 shows that when a rocket is ascending through the atmosphere, a high thrust force is needed to combat the negative forces of drag and gravity to improve acceleration. Once a rocket has left the atmosphere and begins to conduct in-space operations, the gravity forces are low and drag is near zero. Therefore in-space operations do not require a high thrust to accelerate the vehicle. The primary consideration for in-space operation then becomes efficient use of propellants which is characterized by a high specific impulse.

C. BASIC COMPONENTS

Liquid propellant rocket engines used for upper stage propulsion follow the same overall basic design as the majority of bipropellant engines. Perhaps the best method to demonstrate the major components of a bipropellant upper stage engine is to show a generic schematic, such as Figure 10.

Figure 10 shows a basic gas generator cycle with the vehicle propellant tanks; each major cycle is discussed further in the next section of this chapter (Section D). The tank ullage is defined as the volume above the fuel/oxidizer. "The optimum shape of a propellant tank is spherical because for a given volume it results in a tank with the least weight," (Sutton & Biblarz, 2010). Although the optimum tank design is spherical, cylindrical tanks are often used to fit more propellant within the rocket's shape. Turbopumps are a strong preference for upper stage LPREs for high performance.

Liquid propulsion systems utilizing a turbopump are lighter weight and provide the optimum performance for high-thrust, long burn time applications (Sutton & Biblarz, 2010).

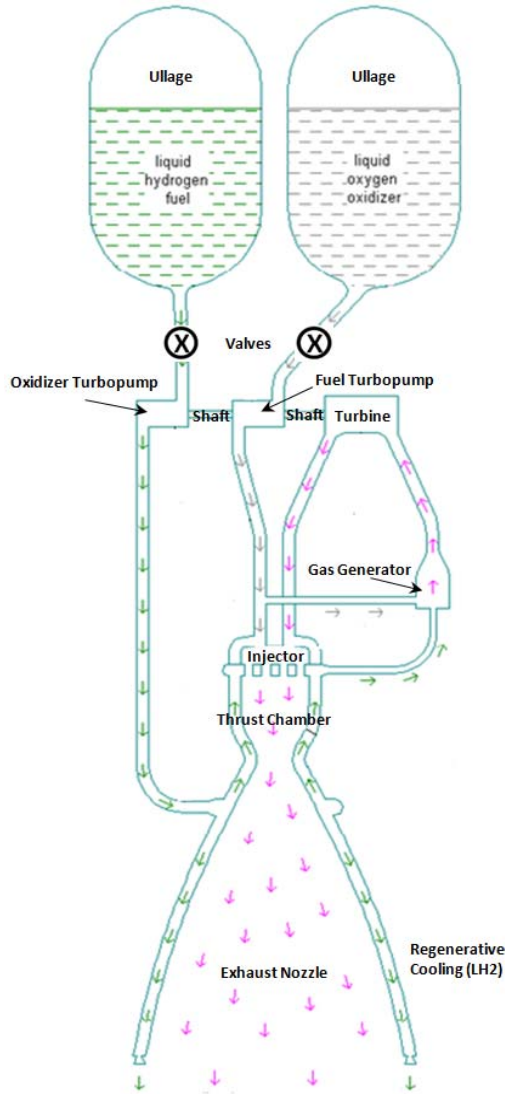

Figure 10. A Schematic of a Bipropellant LPRE Using a Gas Generator (GG) Cycle (After: Kim, 2010)

D. ENGINE CYCLES

Liquid propellant rocket engines with turbopumps are often classified by their engine cycle (Kim, Emdee, & Cohn, 2010). An engine cycle defines propellant flow and the method(s) of gas entrance/exit through the engine turbine(s) (Sutton & Biblarz, 2010).

Engine cycles can be either open or closed. Open cycles exhaust turbine working fluid outside of the engine system. Closed cycles exhaust turbine working fluid into the thrust chamber. Closed cycles have slightly better performance than an open cycle but are more complex. Figure 11 presents a schematic for each of the three major LPRE cycles; the gas generator cycle, the expander cycle, and the staged-combustion cycle.

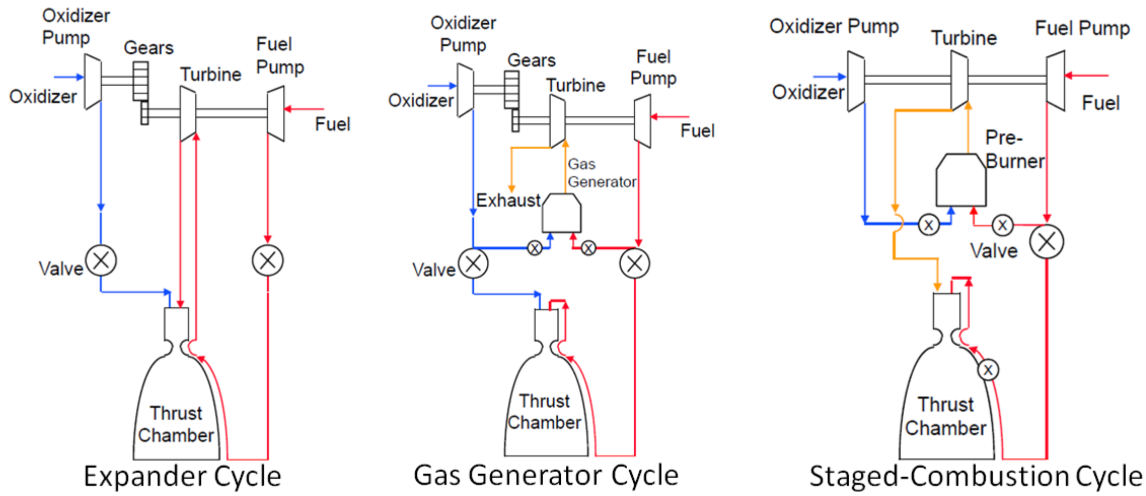

Figure 11. Basic Schematics of the Expander Cycle, Gas Generator Cycle, and the Staged-Combustion Cycle (From: Kim, 2010)

1. Gas Generator Cycle

The gas generator cycle (GG) was the first major cycle to be created. During this cycle there is either a small amount of fuel or oxidizer bled off for use in the gas generator or there is a separate monopropellant fed to the gas generator. The propellant burned in the gas generator creates a gas to drive the turbine of the engine. Turbine exhaust is either discharged overboard or can be injected into the engine's main nozzle. Both methods of exhaust dispersion may create a small additional thrust but at suboptimal *MR* and performance (Sutton & Biblarz, 2010).

2. Expander Cycle

Pratt and Whitney developed the expander cycle in 1960. In the expander cycle heat from the regenerative cooling jacket vaporizes the fuel. After vaporization the fuel is passed through the turbine and into the thrust chamber. Hydrogen is quite frequently used as the fuel for this process, due to its low boiling point. All propellants are burned at an optimal mixture ratio during this cycle (Kim, Emdee, & Cohn, 2010).

3. Staged-Combustion Cycle

The staged-combustion cycle was developed in Russia in 1958. The fuel during this cycle is tapped off similar to the gas generator cycle. Unlike the gas generator cycle, a larger amount of one propellant is used when compared to the other. Depending on which propellant is predominately used the staged-combustion cycle can be either a fuel-rich staged-combustion cycle (FRSC) or a oxidizer-rich staged-combustion cycle (ORSC). The propellants feed into a preburner which burns the propellants creating a hot vapor. This hot vapor is fed through the turbine and, like the expander cycle, into the thrust chamber. All propellants are burned at an optimal mixture ratio during this cycle (Kim, Emdee, & Cohn, 2010).

E. TYPES OF PROPELLANTS

Approximately 170 different liquid propellants have undergone lab testing. This estimate excludes minor changes to a specific propellant such as propellant additives, corrosion inhibitors, or stabilizers. In the U.S. alone at least 25 different propellant combinations have been flown (Sutton, 2003). However, there has not been a completely new propellant used in flight for nearly 30 years (Sutton & Biblarz, 2010).

Many factors go into choosing a propellant for a LPRE. The primary factors include ease of operation, cost, hazards/environment and performance.

Bipropellants, the focus of this thesis, can be either hypergolic or nonhypergolic. A hypergolic combination of oxidizer and fuel will start to burn upon contact. A nonhypergolic needs an ignition source (Larson & Wertz, 2005).

The upper stage LPRE propellant preference in the U.S. is arguably the bipropellant combination of cryogenic liquid oxygen and hydrogen. This fuel combination yields a high specific impulse. This extra performance typically offsets the fuel’s disadvantage of low density.

Low density of a propellant leads to larger fuel tanks. However, a small increase in specific impulse in an upper stage application can have a significant increase in payload to orbit capability (Sutton, 2003 and Sutton & Biblarz, 2010).

F. UPPER STAGE ENGINE COMPARISON

In order to present the reader with a comparison of upper stages worldwide Table 4, below, is shown. Pay special attention to the specific impulse and thrust values among the LOX/LH2 engines. The highest specific impulse among engines in Table 4 are those engines that use a LOX/LH2 fuel and expander cycle.

Engine name	Vehicle application	Manufacturer	Propellants	Engine cycle	Nozzle expansion ratio	I_{sp}, vacuum (s)	Thrust, vacuum (kN/lbf)
RL10B-2	Delta IV Stage 2	PWR (USA)	LOX/LH2	Expander	285	465	109.87/24 700
RL10A-4-2	Atlas V Stage 2	PWR (USA)	LOX/LH2	Expander	84	451	99.2/22 300
J-2	Saturn V Stage 2	PWR (USA)	LOX/LH2	GG	27.5	421	1023/230 000
Vinci	Ariane 5 Stage 2	SNECMA (France)	LOX/LH2	Expander	240	465	180/40 500
LE-5	H1 Stage 2	Mitsubishi (Japan)	LOX/LH2	Expander	140	449	103.2/23 200
LE-5A	H2-A Stage 2	Mitsubishi (Japan)	LOX/LH2	Expander	130	453	121.9/27 400
LE-5B	H2-A Stage 2	Mitsubishi (Japan)	LOX/LH2	Expander	110	447	137.45/30 900
RD-120	Zenit 2 & 3SL Stage 2	NPO Energomash (Russia)	LOX/Kerosene	ORSC	106	350	833.6/187 400
NK-39	N-1 Stage 3	ND Kuznetsov (Russia)	LOX/Kerosene	ORSC	114	353	402/90 400
NK-43	N-1 Stage 2	ND Kuznetsov (Russia)	LOX/Kerosene	ORSC	70	346	1757/395 000
RD-0110	Soyuz Stage 3	KB Khimav-tomatiki (Russia)	LOX/Kerosene	GG	82.2	326	298/67 000
AJ-10-118K	Delta II Stage 2	Aerojet (USA)	N2O4/Aerozine 50	Pressure Fed	65	319	42.7/9600
LR91-AJ-11	Titan IV Stage 2	Aerojet (USA)	N2O4/Aerozine 50	GG	49	316	467/105 000
YF-22	Long March-3 Stage 2	China Great Wall Industry Corp. (China)	UDMH/N2O4	GG	10	295	720.6/162 000
YF-73	Long March-3 Stage 3	China Great Wall Industry	LOX/LH2	GG	40	425	44/9900
Merlin	Falcon 1 Stage 1	SpaceX (USA)	LOX/Kerosene	GG	14.5	304	615.6/138 400

Table 4. Upper Stage LPRE Comparison (After: Kim, Emdee, & Cohen; and Isaowitz, Hopkins, & Hopkins, 2004)

Performance characteristics of historic/current use engines will be compared in Chapter IV in order to derive relationships needed for optimization.

G. CHAPTER SUMMARY

The U.S. can be extremely proud of the role it has played in the development of liquid propulsion. The U.S. been credited with many firsts in the field of liquid propulsion most notably those of Robert H. Goddard.

Rocket engine comparisons are often done using the values of specific impulse and thrust. The reader will see these as important values in analysis ahead.

Basic components of a bipropellant LPRE include the one or more turbopumps, one or more turbines, interconnecting shafts, injectors, one or more trust chambers, and one or more nozzles. LPREs today commonly use regenerative cooling.

The oldest and most common engine cycle used in upper stage liquid propulsion is the gas generator cycle. This cycle differs from the other common cycles in that it is an open cycle. The other two cycles, the staged-combustion and the expander cycles, are more efficient than the gas generator cycle. However, the staged-combustion and expander cycles are also more complex and costly to develop.

There are a large number of different propellant combinations that could be used in a liquid rocket engine. The current preferred propellants used in the U.S. for upper stage LPREs are hydrogen and liquid oxygen. These propellants produce a high specific impulse that caters well to the mission of the upper stage.

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IV. ANALYSIS

The development of evaluation measures and eventual engine selection is a cumbersome task. Although many evaluation measures of a propulsion system can be related to performance, cost, or reliability, the large number of subsystems and interfaces creates a systems engineering (SE) challenge. Since the early development of liquid propulsion, both the concept of systems engineering and development of computer-aided designs have helped drastically to progress liquid propulsion rocket engine technology. "Only through careful analyses and systems engineering studies is it possible to find compromises that allow all subsystems to operate satisfactorily and be in harmony with each other," (Sutton & Biblarz, 2010). This chapter outlines the systems engineering processes for new upper stage liquid propulsion selection, evaluation measures development, and future testing. The focus of the analysis was on developing top-level evaluation measures for the LPRE. In addition, showing where the evaluation measures development fits in the overall process is outlined. Take a careful note that the perspective of this procurement process is from a government viewpoint.

A. SYSTEMS ENGINEERING PROCESS

Systems engineering has been defined in several different capacities, each pertaining to the area in which the SE is taking place. For our intents and purposes we define systems engineering as "a logical process of activities, analyses, and engineering designs, that transforms a set of evaluation measures arising from specific mission objectives in an optimum way. It ensures that all the likely aspect of a project or engineering system have been considered and integrated into a consistent whole," (Sutton & Biblarz, 2010).

For the selection and acquisition of an upper stage propulsion system a waterfall model is recommended, refer to Figure 12 below. The waterfall method, which is commonly used in software development, has a structured multiphase downward progression. Key in this method is the completion of one phase before the undertaking of the next. In the pursuit of eventual engine selection it is important, especially considering the complex integration involved, to follow a systematic acquisition approach such as shown in Figure 12. Note a possible drawback to using the SE waterfall method is the often mandatory completion of one step before another. This thesis fits into the third stage of the waterfall model here, because the research aims to present top level performance evaluation measures.

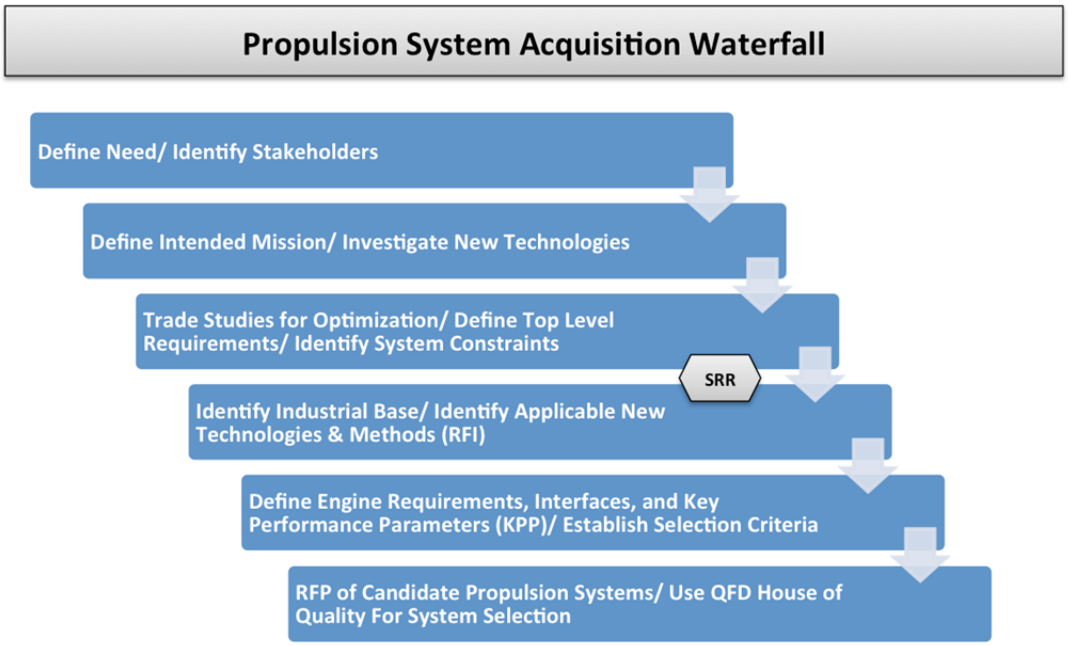


Figure 12. Government Propulsion System Acquisition Waterfall Model

The primary stakeholders identified for the NGE are the Launch and Range Systems Directorate (LRSD), NASA, the Office of Space Launch (OSL) and all of LRSD's primary customers. LRSD's primary customers include Defense Weather Systems Directorate, Space-Based Infrared Systems Directorate, Global Positioning Systems Directorate, MILSATCOM Systems Directorate, and the U.S Navy. It is a recommendation of the author for the systems engineer to explore other possible primary stakeholders. For the purposes of thesis, the national launch forecast (NLF) was used in the creation of a customer population sample. The NLF gives predictions for manifested missions from the present to FY21. The NLF is for government use only. A NLF can be obtained by contacting the Launch and Range Systems Directorate.

As stated earlier in this paper, a systems engineering process should be comprehensive. The systems engineer when using SE methods of optimization and evaluation will consider performance, reliability, and cost. Performance values recommended in this thesis are based on vehicle performance data and the associated research. Although the waterfall model present here will not be discussed at length, it is important to note the meaning of the abbreviations included therein. The Systems Requirements Review (SRR) is primarily used to check the thoroughness/completion of the identification of top-level requirements and system constraints. This review is accomplished before a Request for Information (RFI) is distributed to the public. Note, acquisition programs may have one to several RFIs distributed.

B. EVALUATION MEASURE DEVELOPMENT/SYSTEM CONSTRAINTS

The basis of propulsion system evaluation measures development starts first and foremost with mission requirements. Mission requirements are mapped from operational needs that include application, flight orbit, flight path, flight maneuvers, launch sites, and reliability (Sutton & Biblarz, 2010). Specific propulsion evaluation measures may include thrust, specific impulse, number of restarts, likely engine cycle, likely propellants, engine mass, engine size, mixture ratio, and number of thrust chambers. With a focus on EELV, many top-level evaluation measures are already known or can be assumed rather quickly. This thesis specifically attempts to focus the evaluation measures of thrust and specific impulse in order to achieve the greatest payload increase fleet wide.

1. Thrust and Specific Impulse Development

The optimization of thrust and specific impulse are critical in the evolutionary improvement of liquid propulsion. See the Radial Venn diagram in, Figure 13, as an illustration of the typical performance improvements made in propulsion throughout timeframe.

The purpose of the Radial Venn is to show that the major methods of evolutionary improvement relate to one another. Also notice that with vehicle size constraints, changing the thrust and specific impulse are the only two sound choices for further improvement of design.

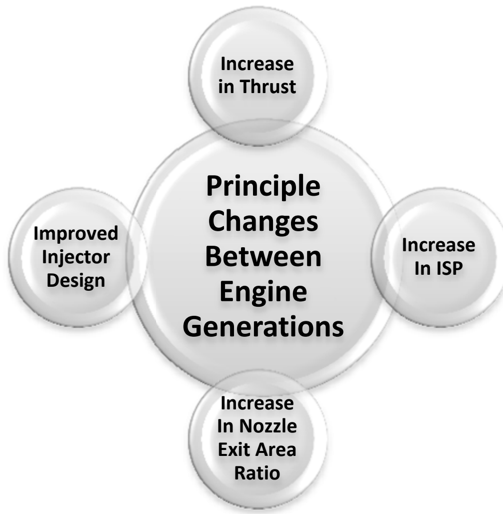


Figure 13. Radial Venn of the Performance Improvements Among Engine Generations

The Next Generation Engine development process must be an improvement over the RL10, an engine that has had 50 years of incremental development and improvement. In optimizing the thrust and specific impulse of EELV, a more focused improvement can be proposed for the next upper stage engine.

Increasing the thrust of a liquid propulsion engine, as seen by Equation 3-14, has a primary effect of reducing vehicle gravitational loss and drag while operating sub-orbitally. However, the higher forces created by the increased thrust may require a more robust vehicle structure. An increase in thrust may also lead to an increase in propulsion system mass unless the engine T/W is improved as well (Sutton & Biblarz, 2010). Thrust increases, in relation to upper stage engine use, will be useful on EELV missions which spend a relatively large fraction of operational time sub-orbital. However, in other EELV upper stage applications an increase in thrust may not impact overall mission performance much at all; this occurs when the upper stage has its primary function while the vehicle is orbital. In other words the thrust force does not need to be larger when the force of drag goes to zero and the force of gravity is small, again, refer to Equation 3-14. For these in-space applications, an increase in engine specific impulse is favored over thrust.

An increase in engine efficiency, or specific impulse, often has a direct effect on upper stage flight performance. Increases in specific impulse are commonly made by an increase in nozzle size. A tradeoff occurs as the increase in nozzle size increases the engine weight. An increase in engine weight has a near pound for pound decrease in payload capability as the engine is carried to final orbit on EELV.

2. Identified System Constraints

Refer to Table 5 for major EELV systems constraints considered during the analysis of performance evaluation measures.

Major Vehicle Constraints-EELV

Constraint	Description
Size/Physical Dimensions	*--New propulsion engine must fit into existing vehicle volume*
Fuel	*--A Hydrogen/Liquid Oxygen must be used to utilize existing vehicle design*
35000 lbf of Thrust or Less	*--Existing Delta IV structure cannot withstand above a thrust of approx. 35000 lbf*
Mixture Ratio	*--Current tank size on EELV limits MR* *--Current tank size on EELV limits burn time*

Major EELV Vehicle Constraints that Affect this Analysis

C. MISSION PERFORMANCE TRADEOFFS

Where should a systems engineer begin to develop evaluation measures? This thesis attempts to answer this question by developing relationships among performance characteristics in order to recommend an optimal specific impulse and thrust combination. Although these performance characteristics will help the systems engineer to define a need, these two performance characteristics alone will not cover all the evaluation measures the systems engineer will need to define.

Top-level evaluation measures may include but are not limited to specific impulse, thrust, thrust to weight ratio, number of starts, nozzle dimensions, life expectancy, mixture ratio, and expansion ratio.

The majority of this thesis section will cover the analysis process that was used in order to develop an optimal thrust and specific impulse that is EELV mission specific.

Specific impulse and thrust are independent evaluation measures for the NGE. They both play a vital role in the overall goal to develop the amount of payload that EELV can place on orbit. As stated in Chapter III, thrust performance is often combined with engine weight to create the thrust to weight ratio; a normalized measurement of the engine thrust. Specific impulse is commonly increased by increasing the nozzle expansion ratio. This will increase engine weight and decrease the thrust/weight ratio. In order to incorporate the engine weight increase associated with an increased expansion ratio, a specific impulse and engine thrust/weight relationship was developed.

Characteristics from upper stage engines worldwide were collected in order to develop a relationship between specific impulse and thrust/weight. As explained in Chapter III, Section D, each cycle may have different performance characteristics. Reiterating Chapter III, Section E, changes among propellant may change the specific impulse values. Furthermore, the NGE will be constrained to the current EELV plumbing, which was designed for the RL10 LH2/LOX propellant combination. Therefore, to keep the specific impulse and T/W relationship in family, upper stage engines using an expander cycle and a LH2/LOX propellant were used for analysis. Thrust and specific impulse were not comparable among these engines due to the performance of each engine being based on a different *MR*. Using the mixture ratio relationship in Figure 14, polynomial curve fits for specific impulse and thrust as a function of *MR* were used to normalize the data at a mixture ratio of 5.88. Normalizing thrust and Isp to a single *MR* is important, because eq 3-10 and eq 3-13 show that these parameters are a function of *MR*. This normalization would approximate the values of each engine to what they would be if the engine were to be running at a mixture ratio of 5.88. As the reader may recall from Chapter III, a mixture ratio of 5.88 is the *MR* that the current RL10B-2 uses and this is also the *MR* the program selected for evaluating NGE engine capabilities to be compatible with current propellant tank designs. Figure 14 shows the polynomial trends that were identified.

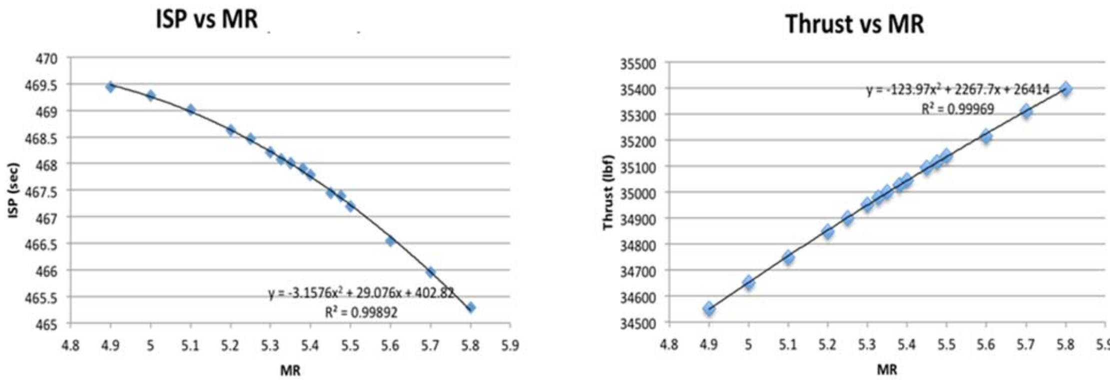


Figure 14. Specific Impulse and Thrust as a Function of the Mixture Ratio

After normalizing for mixture ratio, a relationship between specific impulse and T/W could be developed. Many different combinations of engines and curve fits were attempted. A polynomial relationship was seen to be the best relationship. This relationship was developed for the combination of the evolutionary RL10 family of engines, the Vinci engine, and the Japanese made LE-5A engine. This mixture of foreign and domestic engines as well as old/new technology presented a good pool in which to build the relationship. See Figure 15 as an illustration of the polynomial performance characteristics relationship. Notice from Figure 15 there are no significant shifts in the performance values of the Vinci engine, the newest upper stage engine of the world, when compared with the early engines of the RL10 family, developed primarily in the 1960s. Not surprisingly the two most advanced engines, the RL10B-2 and the Vinci, have the highest specific impulse values. However, both of these engines also have the lowest thrust to weight ratio. The lower thrust to weight of these two engines can be attributed to a larger nozzle than the other engines. The larger nozzle increases specific impulse, but also increases the weight.

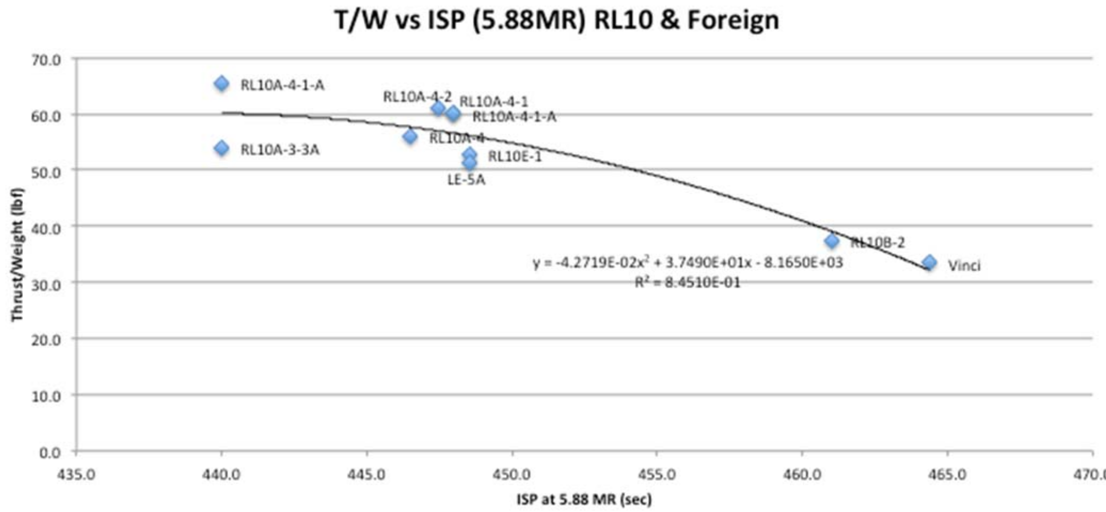


Figure 15. A Polynomial Relationship Between Performance Characteristics

The new relationship created in Figure 15 can lead to preliminary conclusions about the rocket's payload throw weight capability. Recall from Chapter III, Section A. that specific impulse is the measure of thrust per unit flow rate. Increasing the specific impulse by even just a few seconds can drastically increase payload throw weight. Therefore increasing specific impulse will increase the payload throw weight overall, but also have a direct effect on increasing engine weight, suggesting there is an optimum specific impulse to improve rocket performance.

Analysts from the Aerospace Corporation, using in house equipment and software, were able to run mission trajectory simulations while changing the variables of thrust and specific impulse. The effect these variable changes had on payload increases or decreases was calculated within the simulations and presented to this thesis author in reference 20. An example of thesis simulation data that was given is shown in Table 6.

Atlas V 401

Thrust (lbf)	Isp (sec)	P/L Delta from Current Nominal (lb)
Current Nominal 22,233	450.7	
25,000	455	520
	465	1,050
	468	1,200
27,500	455	640
	465	1,170
	468	1,310
30,000	455	700
	465	1,220
	468	1,370
32,500	455	720
	465	1,260
	468	1,400
35,000	455	740
	465	1,270
	468	1,410

Table 5. Simulation Data Example for the Atlas V 401 Configuration, GTO

The example, Table 6, shows the increase in payload capability of a simulated Atlas 401 mission, with the thrust and specific impulse varied but engine weight held constant. The simulation did not account for the varying weight of the rocket engine. Using the new relationship, Figure 15, for thrust/weight and specific impulse, the thesis author was able to change the payload increase data to reflect changing engine weight with specific impulse and thrust.

Regression analysis is a modeling technique used to relate one or more independent variables with one continuous dependent variable. "The goal of regression analysis is to identify a function that describes, as closely as possible, the relationship between these variables so that we can predict what value the dependent variable will assume given specific values for the independent variables," (Ragsdale, 2007). Regression analysis was used to create a relationship that could be developed to find the highest payload increase from any given thrust or specific impulse combination.

The general equation for polynomial regression for one independent variable, X, is given below as Equation 4-1.

$$Y = b_o + b_1 X_1 + b_2 X_2^2 \tag{4-1}$$

For this problem, Y_{pl} is the value given to the dependent variable of payload increase. The symbols b_n are the regression constants. Also, for this problem, there are two independent variables of specific impulse, X_{Isp}, and thrust, X_{th}. The final equation for relating thrust, specific impulse and payload increase now becomes Equation 4-2.

$$Y_{pl} = b_o + b_1 X_{isp} + b_2 X_{isp}^2 + b_3 X_{th} + b_4 X_{th}^2 \tag{4-2}$$

Given the regression equation has now been developed, MS Excel was used to solve for the maximum payload increase value by using GRG Nonlinear methods. An example of the MS Solver functional add-in can be seen as seen as Figure 16.

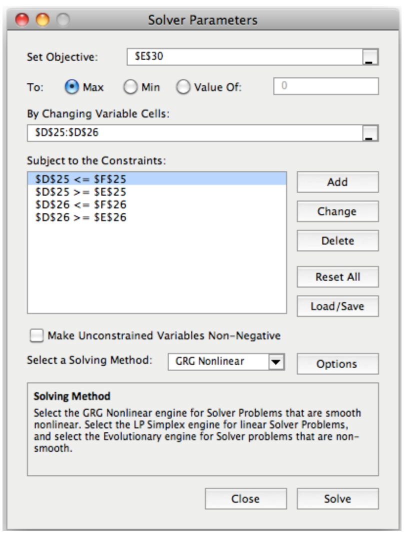


Figure 16. Illustration of the Solver Function

Each mission that the author had simulation data for was developed. The objective was to determine the payload increase by changing the variables of thrust and specific impulse. Constraints of a thrust between 20000–42500lbf and specific impulse between 455–473sec were originally assessed. However, it was noted eventually that the optimum payload increase would remain within these ranges even if the optimization was not constrained.

After optimizing for each Atlas and Delta launch vehicle variant and orbit, a launch vehicle family optimization was performed. However, to develop for each family of vehicles, a weighted factor for the national launch forecast had to be assessed. Table 7 shows the EELV manifest from the present through FY21.

EELV Manifest By Configuration & Orbit
(Present thru FY21)

Delta IV				
Configuration	Orbit	# Manifested	% Manifested	Simulation Data Present
Delta IV M+(4,2)	GTO	1	2.38%	yes
Delta IV M+(5,4)	GTO	6	14.29%	yes
Delta IV M (4,0)	LEO	3	7.14%	yes
Delta IV M+(4,2)	LEO	1	2.38%	yes
Delta IV M+(5,2)	LEO	5	11.90%	
Delta IV M+(5,4)	LEO	0	0.00%	yes
Delta IV Heavy	LEO	3	7.14%	
Delta IV M+(5,2)	HEO	1	2.38%	
Delta IV M+(4,2)	MEO	4	9.52%	yes
Delta IV M+(4,2)	MTO	10	23.81%	
Delta IV M+(4,2)	GEO	4	9.52%	
Delta IV Heavy	GEO	4	9.52%	yes
	Total	42	100.00%	

Atlas V				
Configuration	Orbit	# Manifested	% Manifested	Simulation Data Present
Atlas V 401	GTO	7	16.28%	yes
Atlas V 531	GTO	5	11.63%	
Atlas V 551	GTO	5	11.63%	yes
Atlas V 401	LEO	7	16.28%	yes
Atlas V 541	LEO	1	2.33%	
Atlas V 501	HEO	2	4.65%	
Atlas V 541	HEO	2	4.65%	
Atlas V 411	MTO	8	18.60%	
Atlas V 401	MEO	6	13.95%	yes
	Total	43	100.00%	

Table 6. EELV Manifest, Present–FY21

In looking at Table 7, one can see that there is simulation data missing for a large number of the missions to be flown in the next 10 years. To compensate for missing simulation data, missions with similar orbits were combined. Therefore, the number of manifested missions that made up a manifest weighting factor for GTO, included manifest numbers for GTO, HEO, and MTO (all similar orbits). The combination of each weighting factor covers all launches over a 10 year projection. Reference Table 8 to see orbit weighted factor used. The orbit weighted factor is referenced in the tables below as the percentage manifested.

Delta Orbit Specific Manifest

Oribit	# Manifested	% Manifested
GEO	8	19.05%
GTO	7	16.67%
HEO	1	2.38%
LEO	12	28.57%
MEO	4	9.52%
MTO	10	23.81%
Total:	42	100.00%

Altas Orbit Specific Manifest

Oribit	# Manifested	% Manifested
GTO	17	39.53%
HEO	4	9.30%
LEO	8	18.60%
MEO	6	13.95%
MTO	8	18.60%
Total:	43	100.00%

Table 7. Delta IV and Atlas V Manifested Mission Count

By combining the developed regression analysis from each simulated mission with a manifest weighting factor, the total weighted average payload gain can be calculated from Equation 4-3. This payload gain equation was utilized for both family optimization and eventually EELV fleet optimization.

Payload Gain=(% manifested GEO Missions)x(GEO payload increase)+(% manifested GTO, HEO, MTO Missions)x(GTO payload increase)+(% manifested LEO Missions)x(LEO payload increase)+(% manifested MEO Missions)x(MEO payload increase) (4-3)

As seen in Tables 8 & 9, GTO, LEO and MTO orbits had a significant influence on the equation due to their high mission count.

If multiple configurations had simulation data that had the same orbit, the regression constants were combined to form one set of data for each orbit. Table 9 records the performance optimization of each family using Equation 4-3.

	Delta V Family	Atlas V Family
Thrust (lbf)	34848.5	31492.7
ISP (sec)	463.4	462.3
Payload Increase/per mission	836.5	1266.7

Table 8. Developed Values for Family Performance Evaluation Measures

After assessing each family separately, the EELV fleet was assessed in the same fashion. Using Equation 4-3 the optimum payload increase fleet wide is an average of 998 lbs/per mission with a thrust of 33626 lbf and a specific impulse of 462.9 sec. These developed values for thrust and specific impulse are based on the thrust/weight relationship used earlier as shown in Figure 15. So the 462.9sec specific impulse recommendation is associated with a thrust/weight ratio of 35. To take advantage of any higher specific impulse, the engine must maintain a T/W of 35 or greater.

Careful consideration was utilized to be sure that this optimization for the fleet would not in fact reduce the payload capacity for any one mission. Each mission that was simulated was re-assessed with the EELV fleet developed values, and the results are recorded in Figure 17.

Payload Increase Using EELV Optimized Values

ATLAS V		P/L Increase Using EELV Optimized Values	Revised P/L Increase Using "No Harm" Policy
Configuration	Orbit	Optimal P/L Increase	Optimal P/L Increase
Atlas V 401	GTO	781	796
Atlas V 551	GTO	1201	1219
Atlas V 401	LEO	2594	2603
Atlas V 401	MEO	640	638
DELTA IV			
Configuration	Orbit	Optimal P/L Increase	Optimal P/L Increase
Delta IV M+(4,2)	GTO	350	348
Delta IV M+(5,4)	GTO	1074	1067
Delta IV M+(4,2)	LEO	1974	1904
Delta IV M+(5,4)	LEO	1648	1604
Delta IV M+(4,2)	MEO	62	74
Delta IV Heavy	GEO	-18	0
		*Thrust: 33626 lbf ISP:462.9 sec	*Thrust: 32992 lbf ISP:462.9 sec

Figure 17. Comparison of EELV Config/Orbit Payload Increases Among the Developed EELV Values and the Revised "No Harm" Policy Values

Although the optimization of EELV yielded an average fleet payload increase of 998 lbs/mission, the Delta IV Heavy configuration with a Geosynchronous (GEO) mission using the EELV developed values had a payload decrease of 18lbs. Since a decrease in payload capability is undesirable, a "do no harm" policy was applied to all configurations. The implementation of this policy was to avoid a payload decrease. The regression equation for the Delta IV Heavy GEO orbit was adjusted to find a zero payload decrease by varying thrust and specific impulse independently. When thrust was varied, a thrust value of 32,992 lbf, resulted in a zero payload decrease.

When the specific impulse was varied in the calculations, without changing thrust, it was concluded that the payload would decrease for a higher or lower specific impulse. This is attributed to the original specific impulse and thrust relationship selected in Figure 15. Refer to Figure 18 for a graphical depiction.

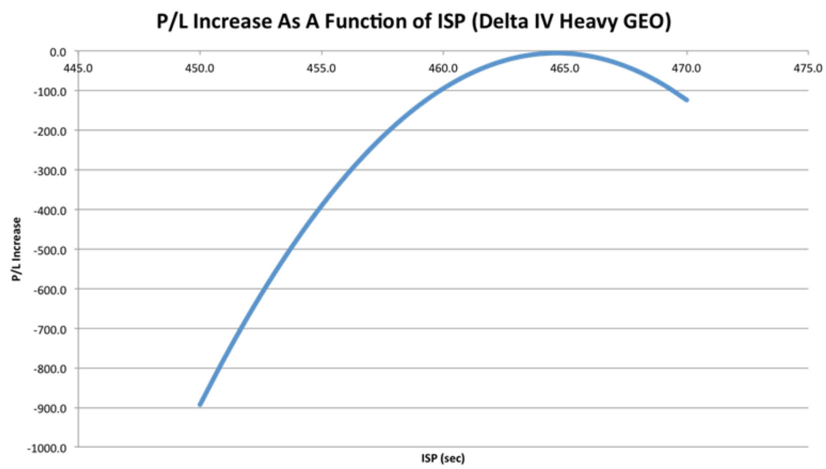


Figure 18. Payload Increase as a Function of Specific Impulse, Delta IV Heavy GEO Config/Orbit

Upon reviewing the results of the analysis completed for the Delta IV Heavy GEO mission, the revised thrust was simulated in the EELV optimization equation to determine the impact. The overall payload capability increase for EELV using a revised thrust of 32992 lbf, resulted in an increase over the current fleet of 995 lbs/mission. This is just 3 lbs less than the calculated developed values for EELV. This revised thrust was used in the equations for each configuration/mission and is presented for comparison in Figure 17. As can be seen, the revised thrust value now concurs with the "do no harm" policy. Graphical depictions of the EELV payload increase as a function of thrust and specific impulse are shown in Figure 19.

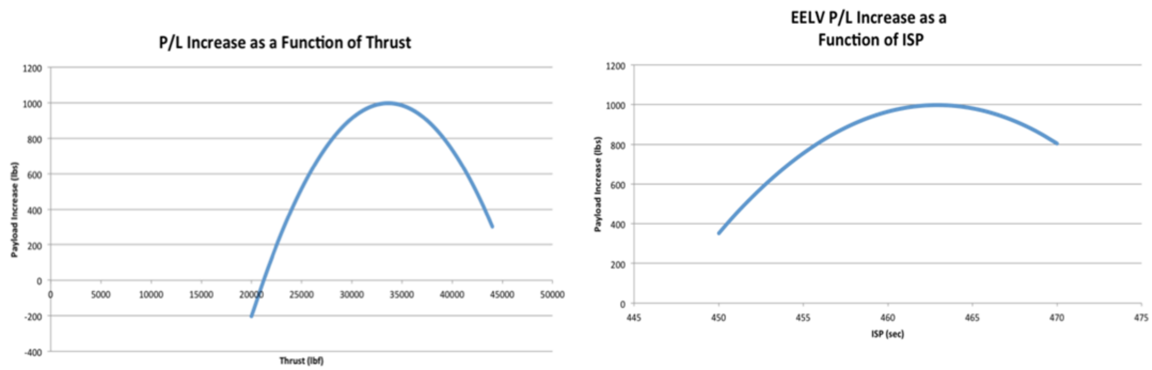


Figure 19. Developed EELV Payload Increase as a Function of Thrust and Specific Impulse

1. Error Calculations

As in any analytical process, there are limits to how well factors can be completely accounted for. This thesis research is no exception, and some areas for further development are identified here:

- **Mixture Ratio Error** - By using a single set of *MR* profiles (Figure 15) as a basis for normalizing engine performance, error will be present on any engine calculation using this relationship. This error association cannot be calculated but is expected to be small compared to the other errors given that eq 3-10 and 3-13 show that thrust and Isp are a function of *MR* for a given chamber pressure and engine geometry.

- **Orbit/Configuration Assumptions -** Many assumptions throughout this work were made in combining like orbits, interpolation of simulated data, or averaging vehicle configuration regression data with the same orbits. This error association cannot be calculated but is believed to be encompassed by the errors below.

- **Engine Weight Calculation Error -** The specific impulse vs. thrust to weight polynomial trend fit equation (Figure 15), which was used in revising the payload increase engine weights, has error associated with it. By comparing the engine weight estimates from this equation to actual engine data, an estimated error bound could be calculated.

- **Regression Analysis Equation Error-** Regression analysis inherently has an associated error. By calculating the difference between the payload increase found using the simulation data and the payload increase found from the regression equation an error estimate was calculated.

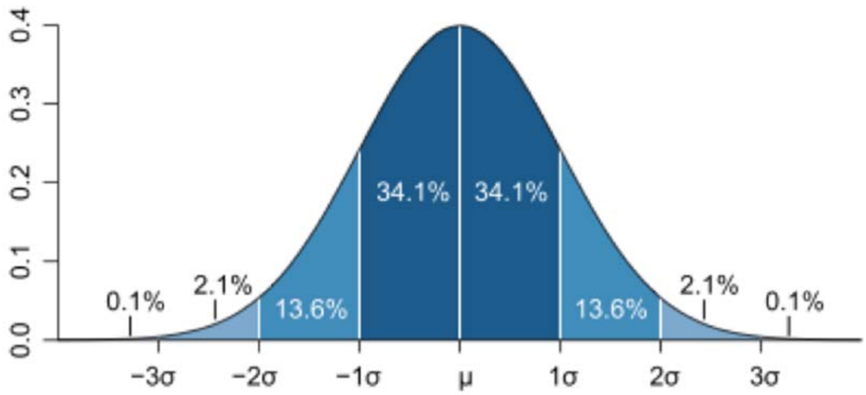
After calculating the error for both the engine weight error and regression error, a sample standard deviation was derived. The general equation for sample standard deviation is

$$\sigma = \sqrt{\frac{\Sigma(x-\overline{x})^2}{N\text{-}1}} \tag{4-4}$$

where σ is the sample standard deviation, x is each value in the sample, $\overline{x}$ is the average of the values, and N is the number of values in the sample. The standard deviation calculations provided a regression error of 35 lbs and an engine weight error of 32 lbs. These forms of error can be combined to give an overall error measurement by using the Root Sum Square (RSS) method. The Root Sum Square equation is as follows

$$RSS = \sqrt{E_{ew}^2 + E_r^2} \tag{4-5}$$

where the E_{ew} is the error associated with engine weight, & E_r is the error weight associated with regression. The final error calculations are shown in Figure 20. The RSS error value is 48 lbs. However, it is a best practice in launch to use a 3-Sigma value error analysis. The total error with 99.8% coverage is 143lbs.



Error	% of Error Covered	Error Calcuation (lbs)
1 sigma	68.2%	48
3 sigma	99.8%	143

Figure 20. Total Calculated Error At 1 Sigma and 3 Sigma

By using the error calculated and the developed values found, a reasonable range can be assessed for the performance evaluation measures in question. Adding an error line as seen on Figure 21 helps us to visualize the possible optimal thrust range.

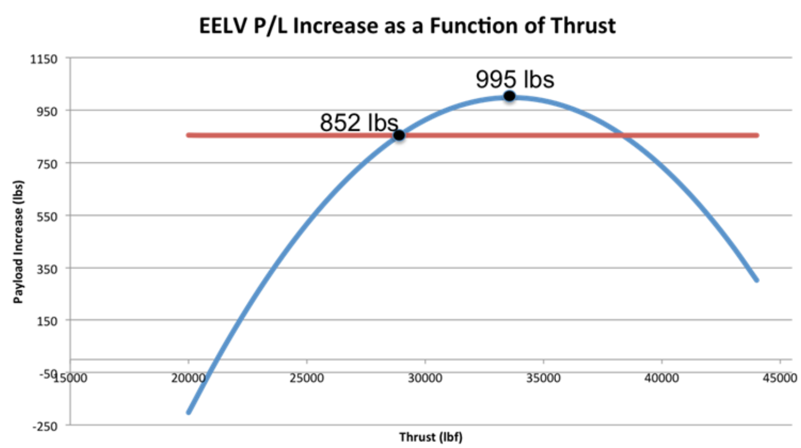


Figure 21. EELV Optimal Payload Increase as a Function of Thrust with Error Range Identified

When looking at Figure 21 a peak value of 995lbs/mission, the optimal EELV payload value, is shown. Given the function is quadratic, any increase or decrease in thrust over the optimal 32992 lbf will have a negative effect on payload increase (with fixed T/W). Using the error calculation of 143lbs at 3 sigma shows the developed payload increase can lie between 852lbs and 995lbs per mission. The associated target thrust, with error included, is a range from approximately 29000–33000lbf. Noticeably, this range leaves out the upper error boundary of Figure 21. However, the analysis associated with Figure 17 showed that an increase in thrust resulted in negative Delta IV Heavy GEO performance and therefore would violate the “do no harm” policy and is not an acceptable range for the target thrust. Both upper and lower bounds decrease thrust, so this is not the reason for ignoring the upper bound. The do no harm policy is the reason for using only the lower bound.

Using the same approach identified for finding a thrust range can be used to find a range for specific impulse. However, specific impulse is much more sensitive than thrust. Adding a range to specific impulse using the error calculation gives a specific impulse range that drastically affects the optimization results. Therefore, it is the recommendation of this author to use the EELV developed specific impulse value of 463 sec or higher as a NGE performance requirement, while keeping the engine thrust to weight at 35 or higher.

D. ENGINE CYCLE TRADEOFFS FOR UPPER STAGES

Deciding on an engine cycle is one of the foremost steps to engine selection. In developing performance evaluation measures, this thesis has worked exclusively with data from expander cycle second stage engines. This, however, should not detract from the feasibility of other engine cycles. A tradeoffs comparison of engine cycles is illustrated in Table 10.

Cycle Type	Gas Generator (GG) (Open)	Expander (Closed)	Staged-Combustion (Closed)
Advantages:	-simple -lower pressures -lower inert mass -lower development cost	-good specific impulse -fair engine simplicity -no gas generator -possible smaller vehicle	-best specific impulse -smaller thrust chamber size -possible smaller vehicle
Disadvantages:	-slightly lower specific impulse	-heavier engine -more expensive -increased complexity -heat transfer to the fuel limits available power to the turbine	-more expensive -increased complexity

Table 9. Tradeoff Comparison among Popular Engine Cycles

Because of the high specific impulse and potential for smaller size (light weight), the staged-combustion cycle should definitely be considered for use as the second stage engine cycle. Chapter V in this thesis will suggest further the areas of study with the stage-combustion cycle in order to reasonably consider its potential benefits.

E. TEST EVALUATION MEASURES FOR ENGINE QUALIFICATION

The foundational goals of a rocket propulsion test program is mitigation of the extreme risk inherent to space flight, verification of system performance evaluation measures, and verification of functional objectives (Rahman & Hebert, 2005). Over the last 60 years liquid propulsion testing has changed significantly worldwide (Emdee, 2001). The primary change has been the vast use of computers in simulation, testing, and design. Still today with the uniqueness of each rocket development program and with new technology emerging rapidly, it is difficult to assess what is needed in a test program. Furthermore, historical data shows that testing and testing hardware still account for approximately 75% of the total engine development costs ("Test and evaluation," 2011).

Like the recommended acquisition process a waterfall model was developed for testing, Figure 22.

The Preliminary Design Review (PDR) is accomplished after the selection of a contractor and has the purpose of giving the government an opportunity to review the contractor's design concept before detailed designs and testing begins. Between SRR and PDR, as stated, is the selection of a contractor/engine concept. It is absolutely vital that experts from the areas of manufacturing, service, materials, stress analysis, and safety are involved in propulsion design selection (Sutton & Biblarz, 2010). The involvement of a variety of experts will ensure an integrated effort and will help to service the systematic waterfall process.

The Critical Design Review (CDR) assesses the detailed designs prior to manufacture. The testing and acquisition processes are assessed on a cost, schedule, and performance basis.

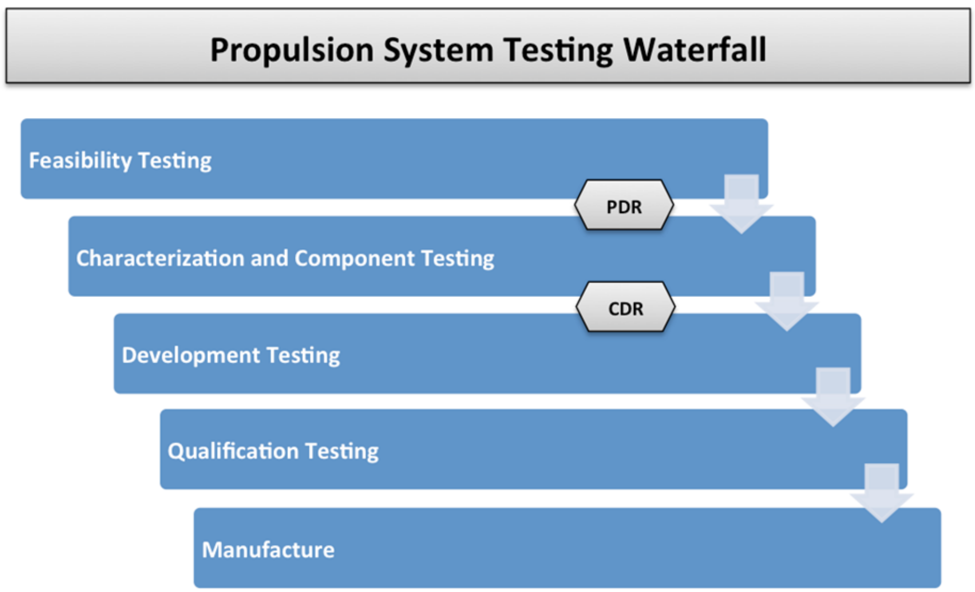


Figure 22. Propulsion System Testing Waterfall Model

As Figure 22 proposes, there are four major types of testing before final qualification. These four phases are Feasibility Testing, Component Testing, Development Testing, and Qualification Testing. Characterization of the system runs concurrently with Component Testing and is done through analysis, modeling and simulation. Test quantity, number of engine fires, and total cumulated firing time performed during development are key factors in reliability estimation and differ among each phase (Pempie & Vernin, 2001). Reference Table 11 for historical data on upper stage engine testing.

Designation	Time from Program Start to Qualification	Engine Life (firings / secs)	Burn Time (secs)	Feasibility			Development			Qualification			Total Development and		
				Engines	Firings	Seconds	Engines	Firings	Seconds	Engines	Firings	Seconds	Engines	Firings	Seconds
HM7A	6 yrs ('73-'79)	-	570	-	-	-	-	-	-	-	-	-	11	-	25,000
HM7B	3 yrs ('80-'83)	-	745	-	-	-	-	-	-	-	-	-	10	-	-
J-2	6 yrs ('60-'66)	30 / 3750	450	-	-	-	36	1,700	116,000	2	30	3,807	38	1,730	120,000
J-2S*	4 yrs ('65-'69)	30 / 3750	450	1	-	10,756	6	273	30,858	Development only			Development only		
LE-5	8 yrs ('77-'85)	-	600	3	54	2,587	5	188	13,414	3	134	14,292	8	322	27,706
LE-5A	5 yrs ('86-'91)	14 / 2920	535	0	0	0	2	66	6,918	2	52	9,238	4	118	16,156
LE-5B	4 yrs ('95-'99)	16 / 2236	534	1	8	237	1	23	1,077	4	79	11,963	5	102	13,040
RL10A-1	3 yrs ('58-'61)	-	380	-	-	-	>230	-	-	-	-	-	>230	707	71,036
RL10A-3-3A	1 yr ('80-'81)	23 / 5800	600	0	0	0	4+	214	18,881	1	24	5,864	5+	238	24,745
RL10A-4	3 yrs ('88-'91)	27 / 4000	400	3+	51	8,321	2+	73	15,055	1	38	5,265	3+	111	20,320
RL10A-4-1	1 yr ('94)	28 / 3480	400	0	0	0	1	5	2,068	1	42	3,683	2	47	5,751
RL10B-2	3 yrs ('95-'98)	15 / 3500	700	1	119	1,701	3+	125	11,605	1	30	4,044	4	155	15,649
YF-73	7 yrs ('76-'83)	-	800	-	-	-	-	-	-	-	-	-	-	120	30,000
YF-75	7 yrs ('86-'93)	-	500	-	-	-	-	-	-	-	-	-	-	-	28,000

Table 10. Upper Stage LOX/LH2 Testing Summary [*J-2S was never qualified] (From: Emdee, 2001)

The main conclusion that can be derived from Table 11 is that there is no identifiable pattern among upper stage engine testing. Considerations should be taken on whether the NGE contractor plans to use some already proven technology or completely new technology. Of course, the less proven the technology used in the engine the more testing it must undergo. A recent example of this is the Vinci engine. The Vinci is the most advanced upper stage engine to date. After reviewing Reference 1 it was observed that the Vinci engine testing, if qualified as planned in 2015, will have undergone ten years of testing. This far exceeds the testing durations of any of the engines included in Table 10 and could be driven more by programmatic budget reasons and not technology challenges. Specific testing numbers, although given in the Reference 1, were still incomplete at this time. Therefore, no direct conclusion can be derived from either comparing the current Vinci testing to previous tests or using the Vinci as a baseline for future upper stage testing.

The Joint Army Navy NASA Air Force (JANNAF) Test Practices and Standards Panel created the *Test and Evaluation Guideline for Liquid Rocket Engines*, Reference 18. This guide's creation uses input and review of 47+ expert contributors. The author fully endorses the usage of this guide as a foundational baseline for LPRE testing. Furthermore, the author's recommendation for NGE testing below comes directly from Reference 18.

Testing Recommendations:

- To remain consistent with the NASA-STD-5012, 6 unique engine samples holding similarity with the flight design are required for verification.
- It is recommended that there be at least 2 qualification engine samples.
- Very low pressure and cold propellants generally require more available energy to ignite, so vacuum testing under conditioned environments is recommended for upper stage engines.
- It is recommended that at least one integrated propulsion system sample be tested. The intent is to replicate all the major functions of the integrated propulsion system to capture integrated propulsion system interactions.
- Utilizing this information for test planning, it is recommended that the qualification portion of the test program not be initiated until the latter portion of the total test program. There tends to be efforts to reduce test program durations by running engines in parallel on multiple facilities. This is viable, but requiring a development test program to complete at least 80% of its tests is a good rule before starting qualification testing.

Recommendations above assume a new engine system with no design heritage.

1. Test-Like-You-Fly

One method of testing that has consistently worked well for the government is test-like-you-fly. This type of testing encompasses as much of the operational worse case conditions as possible. This section of the thesis will inform the reader of the test-like-you-fly methodology and present the relationships between this methodology and EELV.

The overarching principle of test-like-you-fly is testing each piece of flight hardware and every different vehicle configuration in the same conditions as the vehicle or hardware would experience in flight. In cases where expected flight conditions cannot be replicated for testing, larger margins are added for reliability ("Test and evaluation," 2011) .

The test-like-you-fly is not new to the aviation world. However, this testing concept has been the staple of EELV reliability and mission assurance since EELV conception. This specific testing methodology has paved the way for the current 100% mission success rate EELV has obtained. The author recommends the continued use of the test-like-you-fly method for future NGE testing.

F. CHAPTER SUMMARY

This chapter has introduced baseline methods for system acquisition and testing. This chapter has also developed the tradeoffs among performance factors in liquid propulsion and rocket engine cycles. The main objective of this thesis was met as a mission performance equation was developed through regression, trending, and GRG Non-linear solving to determine optimum performance design factors specific to EELV historical mission profile data.

Using a systems engineering waterfall method, a foundation for the second stage Next Generation Engine acquisition and testing has been suggested.

The author, through evaluation measures development using analytical relationships and regression methods, recommends a specific impulse of 463sec or higher, while keeping the engine thrust to weight at 35 or better. Using the same methods, the author recommends a thrust value of 29000–33000lbf.

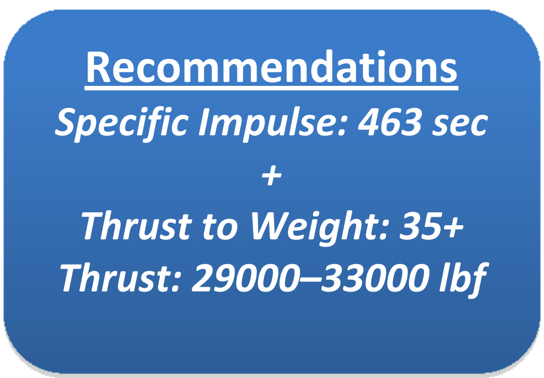


Figure 23. Author Recommendations for Performance Measures

Although each engine cycle presents its own benefits, the expander cycle has been the primary choice for second stage liquid propulsion worldwide. With consideration given to current vehicle hardware and the history of expander usage throughout the life of EELV, the author has chosen to run analysis for an expander cycle NGE. An area of further research not covered but may be of interest, would be to rerun the thesis analysis with a staged-combustion engine assumption. The author recommends using a thrust to weight ratio of 45 and 60 for comparison against the current expander cycle data. For government cleared personnel, a copy of the templates and analytical spreadsheets used for this thesis can be obtained by contacting the author at jason.panczenko@yahoo.com.

Test evaluation measures for engine development vary drastically with each engine program. The author endorses the recommendations of the Joint Army Navy NASA Air Force (JANNAF) Test Practices and Standards Panel which produced the *Test and Evaluation Guideline for Liquid Rocket Engines*, Reference 18. The author, in agreement with the JANNAF panel, recommends the following test practices:

- To remain consistent with the NASA-STD-5012, 6 unique engine samples holding similarity with the flight design are required for verification.
- It is recommended that there be at least 2 qualification engine samples.
- Very low pressure and cold propellants generally require more available energy to ignite, so vacuum testing under conditioned environments is recommended for upper stage engines.
- It is recommended that at least one integrated propulsion system sample be tested. The intent is to replicate all the major functions of the integrated propulsion system to capture integrated propulsion system interactions.
- Utilizing this information for test planning, it is recommended that the qualification portion of the test program not be initiated until the latter portion of the total test program. There tends to be efforts to reduce test program durations by running engines in parallel on multiple facilities. This is viable, but requiring a development test program to complete at least 80% of its tests is a good rule before starting qualification testing.

Considering the current reliability success of EELV, the author recommends the continuation of the test-like-you-fly method for the next LPRE second stage follow on.

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V. CONCLUSIONS

A. CONCLUSIONS

United States' liquid rocket propulsion technology today is based on commercial need, DoD budgets, and the DoD policy of assured national space access. The current solution to U.S. space lift need is the EELV systems, the Atlas V and Delta IV. Both EELV systems today use variants of the RL10 liquid propulsion engine. The RL10 engine has been in service for over 50 years. Due to the long service life of the RL10 it is has issues of with parts obsolescence, low rate production, and escalating cost growth.

Basic components of a bipropellant LPRE include the one or more turbo-pumps, one or more turbines, interconnecting shafts, injectors, one or more thrust chambers, and one or more nozzles. LPREs today commonly use regenerative cooling. Each engine cycle has its benefits. When compared to the gas generator cycle, the staged-combustion and the expander cycles generally produce higher efficiencies. Due to their complexity, the staged-combustion and expander cycles, however, typically have a higher cost and longer lead time associated with them. With research considerations for both EELV structure design and historical second stage cycle usage, the author chose to use the expander cycle as the NGE cycle of choice.

This thesis research indicated that there are several different propellant combinations that could be used in a liquid rocket engine. The current preferred propellants used for EELV LPREs are hydrogen and liquid oxygen. These propellants produce a high specific impulse that caters well to the mission of the upper stage. Hydrogen and liquid oxygen were the author's choice in this thesis, due to EELV being constrained in plumbing and infrastructure design. Evaluation measures development using analytical relationships and regression methods found an optimal specific impulse of 463sec and thrust of 32992 lbs for EELV.

As propulsion system acquisition and testing are very sequential, a systems engineering waterfall method was recommended and developed to frame this analysis. Primary stakeholders cited by the author included the Launch and Range Systems

Directorate (LRSD), NASA, the Office of Space Launch (OSL) and all of LRSD's primary customers. LRSD's primary customers include Defense Weather Systems Directorate, Space-Based Infrared Systems Directorate, Global Positioning Systems Directorate, MILSATCOM Systems Directorate, and the U.S Navy. Because identifying stakeholders may be a reiterative process early in acquisition, the Air Force systems engineer would need to continue to reevaluate stakeholders early in the acquisition waterfall process.

B. RECOMMENDATIONS

From the analysis conducted in this thesis, performance evaluation measures of 463sec or higher specific impulse and 29000–33000lbf thrust for the NGE should be used while maintaining an engine thrust to weight ratio of 35 or higher. Although test programs for engine qualification differ significantly, a test approach which mirrors the suggestions in the JANNAF panel in Reference 18, *Test and Evaluation Guideline for Liquid Rocket Engines* is advised.

1) To remain consistent with the NASA-STD-5012, 6 unique engine samples holding similarity with the flight design are required for verification.
2) There should be at least 2 qualification engine samples.
3) Because very low pressure and cold propellants generally require more available energy to ignite, vacuum testing under conditioned environments should be conducted for upper stage engines.
4) At least one integrated propulsion system sample should be tested, as the intent is to replicate all the major functions of the integrated propulsion system, in order to capture integrated propulsion system interactions.
5) To use this information for test planning, the qualification portion of the test program should not be initiated until the latter portion of the total test program. (There tends to be efforts to reduce test program durations by running engines in parallel on multiple facilities. While this is viable, requiring a development test program to complete at least 80% of its tests is a good rule before starting qualification testing.)

Finally, based on the success of the EELV program and successful qualification process already in place, the author recommends the continuation of the test-like-you-fly method. The test-like you-fly method has been the foundation of mission success for over a decade.

C. AREAS FOR FUTURE RESEARCH

After the NGE design is finalized, a future investigation into the vehicle lift-off thrust–to-weight ratio may be critical to understanding NGE weight effects. Such an investigation may lead into further analytical work on how a heavier upper stage engine may affect Delta IV Heavy LEO missions, which are lift-off-thrust-to-weight limited. The possibility that this new upper stage engine becomes human rated in support of NASA missions is an additional factor to consider.

Research into the potential performance of a staged-combustion second stage engine is an area for research, as would be varying the thrust-to-weight ratio in the above calculations for a performance comparison to the current research values. A baseline thrust to weight ratio of 45 and 60 would provide a good starting point for cycle performance comparisons.

In addition, experimentation with different propellants for a future second stage, and research into new propellant technology information is available in Reference 26. While only liquid propulsion second stage engines have been considered here, hybrid engine design is progressing. Although this technology was underdeveloped at the time of this research effort, it may be a viable option for LPREs after the NGE.

While the purpose of this thesis was to identify performance requirements for the NGE, reliability and cost are also both important factors to the SE process. The performance recommendations presented here could be extended to consider both cost and reliability evaluation measures. Reference 29 pertains directly to reliability, and could be compared with the results contained in this thesis. Finally, an Air Force systems engineer, if working cooperatively with NASA, would be able to explore the extensive testing requirements for human rating a propulsion system. For human rating of EELV, Reference 27 could be consulted.

This thesis has presented a comprehensive investigation into specific values for the performance the Next Generation Engine should obtain. Although not all encompassing, the studies and presentation therein gives the foundation for future DoD liquid propulsion procurement. It is the hope of the author that the recommendations in this thesis be followed in order to procure successfully a follow on second stage engine for the future of space lift in the United States.

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